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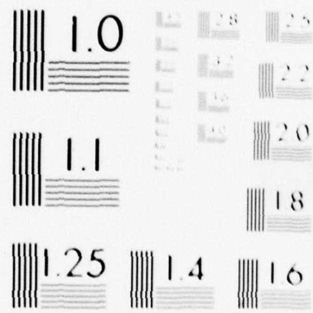
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LEVEL II

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WINDSHIELD TECHNOLOGY DEMONSTRATOR PROGRAM-CANOPY DETAIL DESIGN OPTIONS STUDY

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SEPTEMBER 1978

TECHNICAL REPORT AFFDL-TR-78-114

Final Report Period February 1977-September 1978

AIR FORCE FLIGHT DYNAMICS LABORATORY
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This report has been reviewed by the Information Office (OI) and is releasable to the National Technical Information Service (NTIS). At NTIS, it will be available to the general public, including foreign nations.

This technical report has been reviewed and is approved for publication.

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20. ABSTRACT (Continue on reverse side if necessary and identify by block number)

This report documents the work accomplished for the Windshield Technology Demonstrator Program. The studies, analyses, testing, and development accomplished during this program involved a total system approach required for aircraft canopies in the context of the continuing Air Force generic windshield development programs. State-of-the-art applications of new transparency materials have been devised from both military and commercial aircraft with major attention directed to the topics of bird impact resistance, structural design integration, systems integration, and design for maintainability and

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Qualification Testing
Radar Cross Section (RCS)
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Static Structural Analysis
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Transparency

20. ABSTRACT, (Continued)

reliability.

The authors and Air Force Flight Dynamics Laboratory (FEW) agree that the various disciplines and essential technical design concepts represented, including associated reports noted in Section IV, should all be utilized in arriving at an optimum design of the canopy system for any production aircraft.

FOREWORD

This report is one in a series of reports that describes work accomplished by Douglas Aircraft Company, McDonnell Douglas Corporation, 3855 Lakewood Boulevard, Long Beach, California 90846, under the Windshield Technology Demonstrator Program. This work was sponsored by the U.S. Air Force Flight Dynamics Laboratory, Wright-Patterson Air Force Base, under Contract F33615-75-C-3105, Project 2202-02-01.

Lieutenant L. Moosman (AFFDL/FEW) was the Air Force Project Manager who monitored the program and provided reference material in a timely manner.

First Lieutenant J. Hager (ASD/YDEF) of the F-16 SPO provided extensive coordination and General Dynamics generated data in a timely manner.

Under separate contract to the F-16 SPO, General Dynamics, Fort Worth, Texas, supplied applicable F-16 engineering data and drawings in a timely manner.

Mr. J. H. Lawrence, Jr. was Program Director for the Douglas Aircraft Company. Principal investigators and co-authors were:

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This report was initially submitted to the Air Force in June 1978, and covers work performed during the period February 1977 through September 1978.

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SECTION I
INTRODUCTION

This report reflects the work accomplished as part of the "Windshield Technology Demonstrator Program" under Contract F33615-75-C-3105 in the context of the continuing Air Force generic windshield development programs. This portion of the program was directed toward the study of canopies for single and tandem seating fighter type aircraft with specific application to the F-16 aircraft. State-of-the-art applications of new windshield materials have been derived from both military and commercial aircraft, with major attention directed to the topics of bird impact tolerance, aerodynamic heating effects, structural design integration, and design for canopy maintainability and reliability.

Whether you, the reader, agree or disagree with the technical details contained herein, the authors and the Air Force Flight Dynamics Laboratory (FEW) are in full agreement that the various disciplines and essential technical design concepts represented in this, and in the associated reports noted in Section IV, should all be utilized in arriving at an optimally designed windshield during the Detail Design Phase of an aircraft. It should be emphasized that the external lofted shape for the vehicle used as the demonstrator for this effort was frozen prior to the initiation of this study effort. This means that, as has so often been done in the past, regardless of what design configuration is selected for qualification testing from a structural standpoint, it is left to the transparency manufacturer to gradually evolve his fabrication techniques so as to provide minimally acceptable optical quality for the production aircraft.

An investigative study was accomplished for the determination of requirements that are applicable to an aircraft windshield/canopy system. General Dynamics specifications and reports, the F-16 SPO supplied data and reports, the Air Force Systems Command (AFSC) Design Handbooks, Military Specifications, Federal Aviation Agency (FAA) Regulations, and commercial aircraft industry reports and standard design practices were reviewed to meet this objective.

Section II presents the requirements that are directly applicable to the design, development and operations of an F-16 canopy system.

Within Section III, an assessment was made of the initial F-16 canopy system relative to the requirements which were established in Section II. The same major technical areas which are listed in Section II are addressed in the F-16 design assessment.

An optimum canopy design for application to future production F-16 aircraft was one of the basic objectives of this program. To this end, a series of tests were required for validation of the existing and proposed candidate designs. Section IV summarizes a series of reports having applicability to the initial research phase of the program and a series of tests that were conducted. These reports are:

- AFFDL-TR-76-75, Effects of Laboratory Simulated Precipitation Static and Swept Stroke Lightning on Aircraft Windshield Subsystems.
- AFFDL-TR-76-114, The Determination of Deflection and Stress Distribution for a Laminated Transparent Beam.
- AFFDL-TR-76-156, Damping Static/Dynamic, and Impact Characteristics of Laminated Beams Typical of Windshield Construction.
- AFFDL-TR-77-92, Evaluation of Windshield Materials Subjected to Simulated Supersonic Flight Environments.
- AFFDL-TR-77-96, Testing for Mechanical Properties of Monolithic and Laminated Polycarbonate Materials.
- AFFDL-TR-77-97, Standardized Windshield Fabrication Specification

The results of thermal studies that were conducted during this program are presented in Section V. These thermal studies included consideration of transient temperatures, temperatures when bird strike might be experienced, canopy edge temperatures, thermal factors related to defogging system performance, and the effects of thermal expansion on a canopy design.

During this program effort, it was determined that mechanical properties for transparency materials, used by designers of windshields/canopies, are not available with the high degree of accuracy that is available for metals. For this effort, both monolithic and laminated canopy designs were investigated. Section VI describes the materials studies and testing that was required to establish both low and high strain rate testing conditions at wide temperature extremes for applicable transparency materials. The materials testing was required to establish material values that can be applied to the design of bird impact resistant canopies to afford the pilot maximum protection over the life span of the canopy.

A series of bird impact tests were conducted on the F-16A canopy at Arnold Engineering Development Center (AEDC) in Tennessee. Section VII presents the test objectives, test description, and results of these series of tests, which were conducted on canopies manufactured from 0.50 inch and 0.62 inch thick coated monolithic polycarbonate materials. This section presents the application of these bird impact test results to predict the transparency penetration velocity and deflection due to the impact of a four-pound bird.

The results of a series of edge joint tests are presented in Section VIII. A variety of laminated specimens were subjected to cyclic load tests and thermal shock tests to verify the structural integrity of the attachment area for a laminated transparency in single shear. Simulated flight conditions were based on a 2000-hour design life at typical high performance aircraft operating temperatures and pressures.

Section IX describes the results of a series of salt abrader tests that were performed on aircraft windshield/canopy transparent materials. Each test specimen was impacted with a controlled blast of salt particles at selected temperatures to simulate an aircraft windshield encounter with minute ice particles in clouds or with dust. These tests were conducted to provide data to be used as a qualitative comparison for an assortment of representative transparent windshield specimens.

In the past the airframe industry has found it necessary to spend millions of dollars to build crew compartments which represent the actual aircraft design, for the sole purpose of bird impact verification testing of windshields and windows. This was necessary because detailed analytical methods were not available. Within the scope of this program an analytical method to define the canopy structural response was formulated. This Bird Impact Math Model was utilized to analyze several test conditions for the F-16 canopy. This work is described in Section X.

Section XI presents a number of design concepts that are defined as potential candidates for the F-16 aircraft canopy to meet various bird impact test levels of 350 through 562 knots, and a design that is applicable to the Air Force Manufacturing Technology Program. The basic configuration presented is a laminated design comprised of an acrylic outer ply and a polycarbonate structural ply coupled by a silicone interlayer. The concepts presented include laminated versions of the current F-16 high visibility design, an openable canopy with an integral bow frame, a fixed windshield, a modified version of the current design that utilized an enlarged transparency, and a modified transparency that provides a flat forward vision area.

SECTION II
ESTABLISHMENT OF TECHNICAL REQUIREMENTS

This section is directed toward the establishment of the technical requirements necessary for the design and development of canopy systems applicable to both single and tandem seating, high speed, high performance, fighter type aircraft.

The sources from which these requirements were derived include:

1. General Dynamics/Air Force requirements for the F-16 aircraft.
2. Interpretation of Air Force expected operational useage and performance envelope for the F-16 aircraft.
3. The Air Force Systems Command (AFSC) Design Handbooks.
4. Generic study programs derived from Air Force, military and commercial aircraft programs.

The purpose for establishing these requirements was to define a baseline for the design of alternate F-16 canopies, and associated subsystems, from which design, development, and testing could evolve into production configurations that would provide low life-cycle costs, principally as the result of high reliability and minimal maintainability problems. The requirements must be applicable to both monolithic and laminated transparencies.

To simplify the presentation of this data, the systems requirements have been segregated into major technical areas for individual review. However, it is fully recognized that these several areas are not independent; consequently, their interrelations are accounted for by design integration and optimization of their many technical parameters. Frequently, in the design of a complex system such as a canopy system, not all requirements can be implemented because of costs, lack of available materials, or lack of importance relative to another required feature.

The major technical requirements addressed are as follows:

VISION/OPTICAL

Vision

Optical

ENVIRONMENTAL DESIGN

Thermal

Rain Removal

Defog System

ELECTRICAL/ELECTROMAGNETIC DESIGN

Triboelectric (P-Static)

Lightning

Radar Reflection Control (RCS)

Electromagnetic Pulse (EMP)

STRUCTURAL DESIGN

Structural

Internal and External Loading

Hail Impact

BIRD IMPACT

Bird Impact

Bird Impact Probability

MAINTENANCE PROVISIONS

Maintainability

VISION/OPTICAL

The requirements for vision and optical quality of a crew compartment transparency are discussed in this section.

Vision

Vision and/or visibility requirements associated with the development of a fighter type crew compartment enclosure must take into account the requirements specified by AFSC Design Handbooks and MIL-STD-850B with limiting constraints imposed by aerodynamic shaping requirements.

Space allocations for equipment such as instruments and escape seats must be given prime consideration in developing the location of the pilot/copilot design eyepoints specified in the specifications noted in MIL-STD-850B, as well as, the positioning of the two man crew in tandem arrangement. The forward pilot must have a minimum straight ahead down vision of 11 degrees and the aft pilot must have a minimum of 5 degrees straight ahead down vision unobstructed by the pilot's seat or headrest.

During the development of the lofted shape of the transparency, head clearance must be provided as directed by AFSC Design Handbook DH 2-2 DN 2A1, Page 5, which specifies that the inner surface of a canopy, to provide head clearance, must not be less than 10 inches from the design eyepoint.

The locations and sizes of posts, fairings, and latching mechanisms relative to the vision obstructions are specified in MIL-STD-850B.

Optical

General Dynamics defined optical requirements for the F-16A/B aircraft forward canopy in their F-16A/B Air Vehicle Prime Item Development Specifications identified as 16PS002 and 16PS005 for monolithic transparencies.

The General Dynamics specifications addressed the primary optical requirements defined as luminous transmittance, haze, angular deviation, scratches and inclusions for monolithic materials.

In the paragraphs that follow Douglas has expanded GD's requirements to include laminated transparencies plus additional performance requirements for subjects such as:

- Double Images
- Reflected Images
- Binocular Vision
- Resolution

Each of the above performance requirements are discussed within the following paragraphs as a basis for recommended requirements.

Luminous Transmission Discussion

The F-16A/B Prime Item Development Specification, identified as General Dynamics Specification 16PS002, specifies that the luminous transmission shall be a minimum of 82 percent when measured normal to the transparency, or 71 percent when measured straight ahead in the installed position.

To establish a common baseline for design comparison purposes, the index of refraction and absorption coefficients of the applicable materials were used to calculate for a monolithic and a laminated canopy the following:

	MONOLITHIC 0.500 INCH POLYCARBONATE (P/C)		LAMINATE 0.125-INCH ACRYLIC 0.080-INCH SILICONE 0.750-INCH P/C	
	NORMAL	INSTALLED	NORMAL	INSTALLED
TRANSMISSION - %	82.22	72.22	80.49	70.45
REFLECTANCE - %	9.59	18.62	8.73	17.45
ABSORPTION - %	8.19	9.16	10.78	12.10

Considering the three ply cross section used for the above calculations with no coatings on any surface, a recommended luminous transmission of 70 percent, as measured straight ahead in the installed position, is achievable along the centerline only. The percent transmission calculated for the installed position was based on an assumed surface installed angle of approximately 30 degrees and no surface curvature. Depending on the method of fabrication and the repeatability of the canopy shape, the percent transmission will change as a function of the surface installed angle. Thus an increase in the surface installed angle will decrease the angle of incidence, lowering the surface reflectance and increasing the percent transmission. A decrease in the surface installed angle will cause the opposite to occur. The double curvature geometry of the F-16A/B transparency will also change the angle of incidence of a given light ray being considered analytically. In addition, the effect of haze, or scatter on the transmission of light will be to further decrease the final percent transmission at the installed angle.

Luminous Transmission Requirements

The following light transmission requirements are recommended for the F-16A/B Windshield Technology Demonstrator Program:

Normal to Surface. Eighty (80) percent minimum over the critical area defined as the gunsight area of $\pm$ 6 degrees in azimuth and 3 degrees up to 15 degrees down in elevation with respect to the pilot's nominal eye position. A minimum of 75 percent shall be maintained over the remainder of the canopy.

Installed Angle. Sixty-five (65) percent minimum measured straight ahead from the pilot's nominal eye position. This requirement must assume that there will be a need to have at least one protective coating applied to the canopy regardless of the material selected.

Haze Discussion

The F-16A/B Air Vehicle Specifications specify that the haze shall not exceed three percent.

Haze is a function of surface quality (surface polish, scratches, cleanliness, etc.), purity of material, viewing angle, thickness, and number of plies in the laminate. These factors have the potential for scattering the light beyond acceptable limits; such scattering is described as "haze".

Haze is determined when a transparent sample is normal to a collimated light beam, using ASTM D1003-52, Federal Test Method Standard No. 406-3022 or using the A.Q.D. method described in Reference 2 of the test method. These methods will not be described herein.

Haze, or scattered light, increases with the angle of incidence and reduces the target/background contrast ratio and hence the target detection range; therefore, haze must be kept to a minimum. Emphasis on the normal angle requirement is essential because of the greater difficulty in measuring haze at high angles of incidence.

A survey of various test reports on haze has exhibited a spread from approximately 1.2 percent for monolithic panels up to approximately 5 percent or more for various combinations of laminated materials and thicknesses. A small number of three ply test specimens have shown a spread of 3.1 to 4.9 percent haze.

Haze Requirements

Normal to Surface. The haze shall be 4 percent maximum measured normal to the surface; the design goal is 3 percent maximum. This is measured over a defined critical area. This value will be a primary quality control requirement.

Installed Angle. No recommendation is made for haze at the installed angle. The Normal-to-Surface value will control the values at other installed angles and is more readily measurable.

Angular Deviation Discussion

Angular deviation is normally caused by unparallel surfaces of transparent material(s). The deviation increases with the index of refraction, with manufactured deviation from surface parallelism, and with angle of incidence. Curved laminates create a geometrical angular deviation that is a function of the radius of curvature, the thickness of laminates, and the number of plies. The geometrical angular deviation will increase with higher angles of incidence. It should also be noted that the geometrical angular deviation will vary with high deflections of the windshield which are caused by pressurization loads.

The F-16A/B Air Vehicle Specifications require that the angular deviation shall not exceed three milliradians (or 10.3 minutes of arc) from a predetermined standard as measured from the design eye position. It is further stated that the aircraft fire control system will compensate for the standard error pattern. The error pattern shall consist of a minimum of 50 data points recorded for each transparency with the transparency in the installed position.

As previously stated, surface non-parallelism or curvature creates angular deviation. Since the F-16A/B canopy has a double curvature shape, all light ray traces to, and forward of, the pilot's design eye position will have an azimuth and elevation geometrical deviation error component.

Light ray traces through a canopy having a double curvature is a three dimensional problem but for simplicity the geometry must be reduced to two dimensional diagrams. Figure 1 illustrates the vertical, or (elevation) and longitudinal geometry of the canopy, pilot's display combining plate and the nominal eye position. This figure shows the

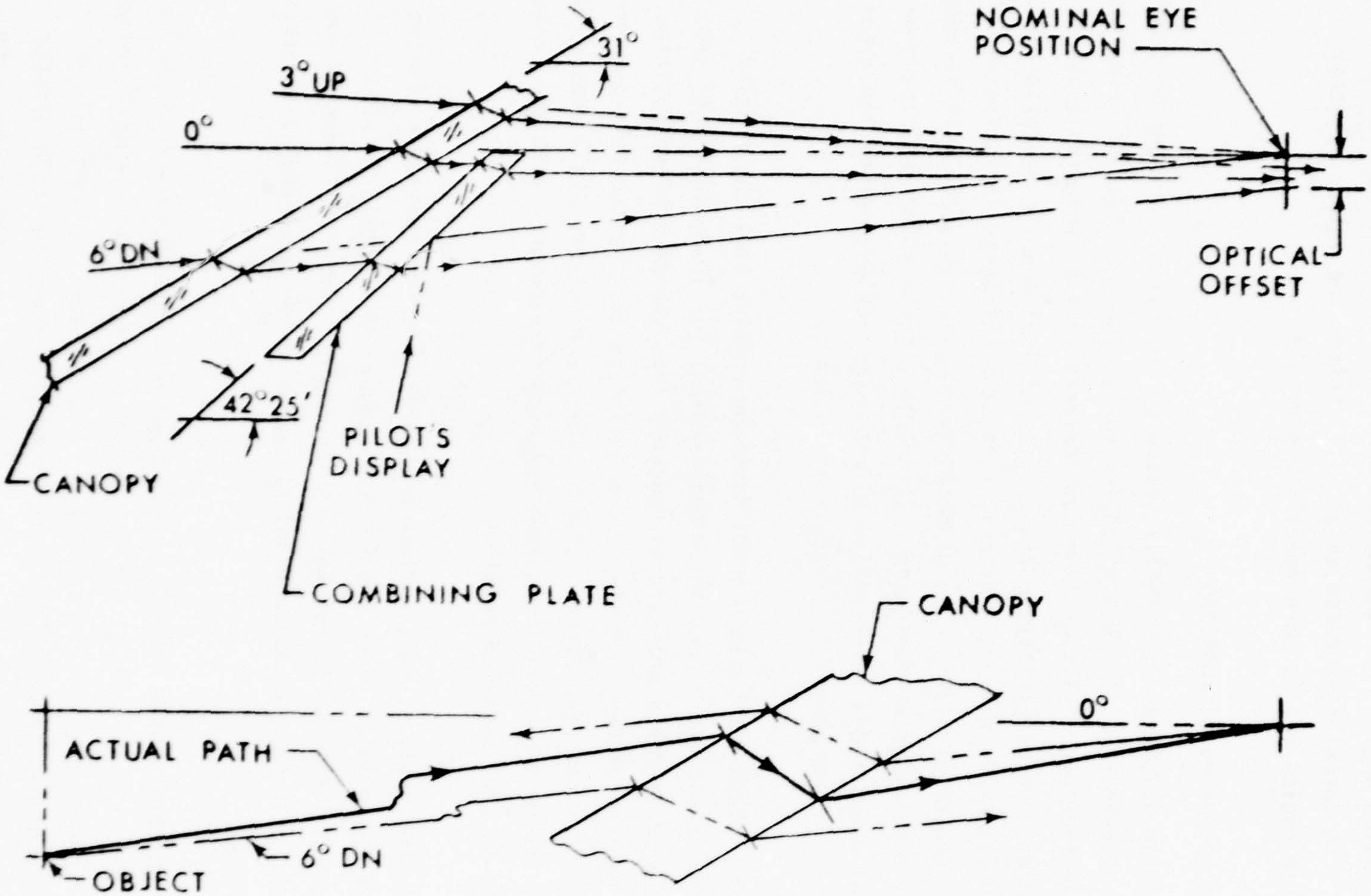


Figure 1. Optical Offset Elevation Angles of 3 Degrees Up, 0 Degrees and 6 Degrees Down.

refraction of three light rays originating from the left, outside the canopy, at true sight angles of 3 degrees up, 0 degrees (horizontal) and 6 degrees down. The true sight rays have intentionally been shown following the law of refraction as the ray goes from left to right. In addition, a reticle ray trace from the pilot's display to the nominal eye position is shown, assuming its generation to be true with respect to the external light ray at 6 degrees down. The optical offset produced at the nominal eye position illustrates and exaggerates the sources of deviation that can result from a change of the shape, thickness, installation angle or material index of refraction. The actual path of a ray from a finite object at some finite distance will not follow the ray traces exactly as shown in the upper diagram of Figure 1.

The lower diagram of Figure 1 illustrates the actual path of a ray trace from an object 6 degrees down, and at some finite distance from the nominal eye position. It is noted that the actual path deviates from the true 6 degrees ray on both sides of the canopy. Theoretically this deviation will approach, but not equal zero, as the object distance approaches infinity, for any ray trace that is not perpendicular to the canopy surface. The insertion of the combining plate into the path of the ray shown will change the value of this deviation. This deviation is of little importance as long as all objects are at infinity, when there is minimal curvature of surfaces, wedginess, or reduction in thickness of material; a head-up-display is not required to compensate for the deviation.

Figure 2 illustrates the geometry of two rays originating from a target essentially at infinity with respect to the parallactic angle required for binocular vision. In addition, two reticle rays from the pilot's display are shown intercepting the left and right eyes. The left hand refracted target ray is shown enlarged for clarity.

The first optical deviation shown is a result of the non-parallelism between the entrance and exit normals. The second optical deviation is

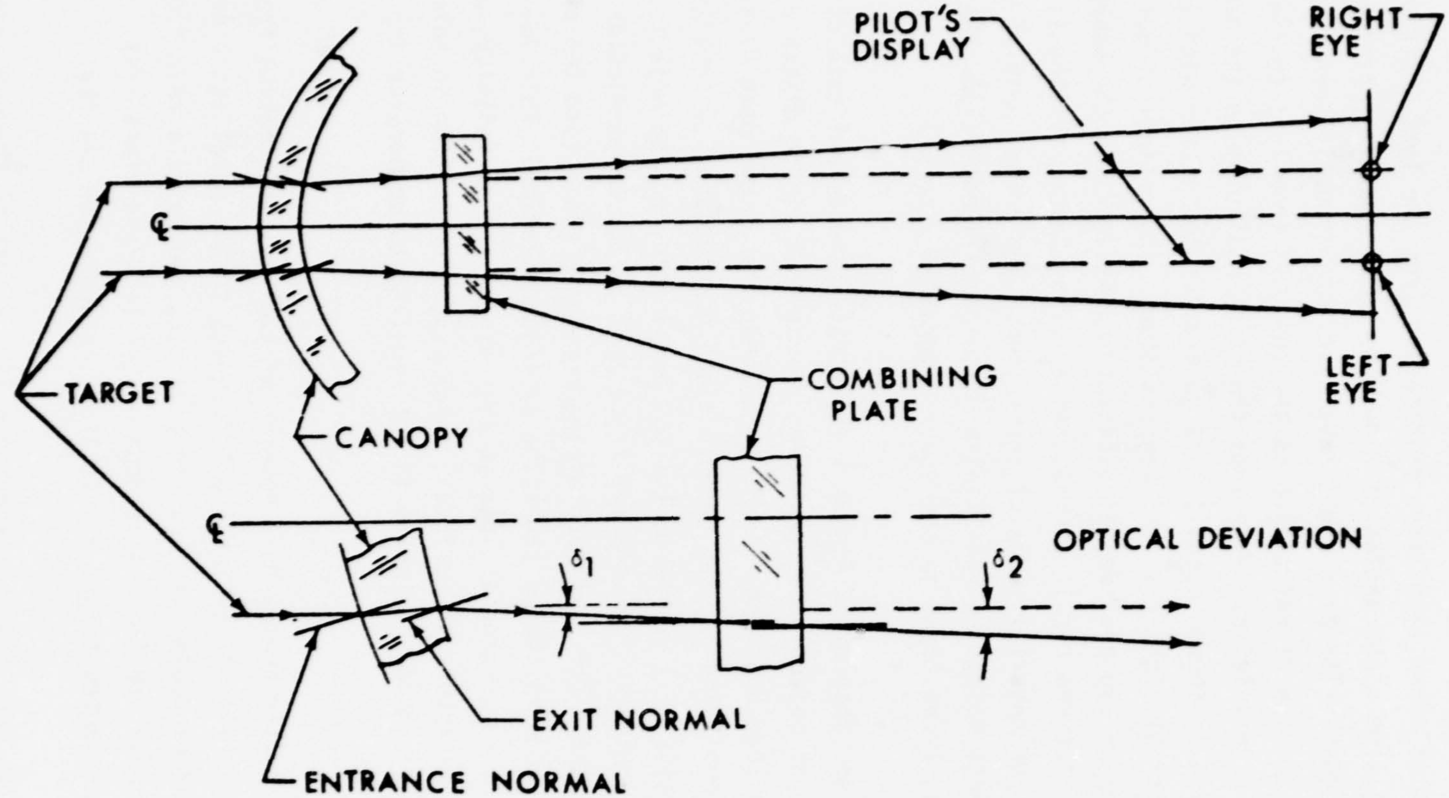
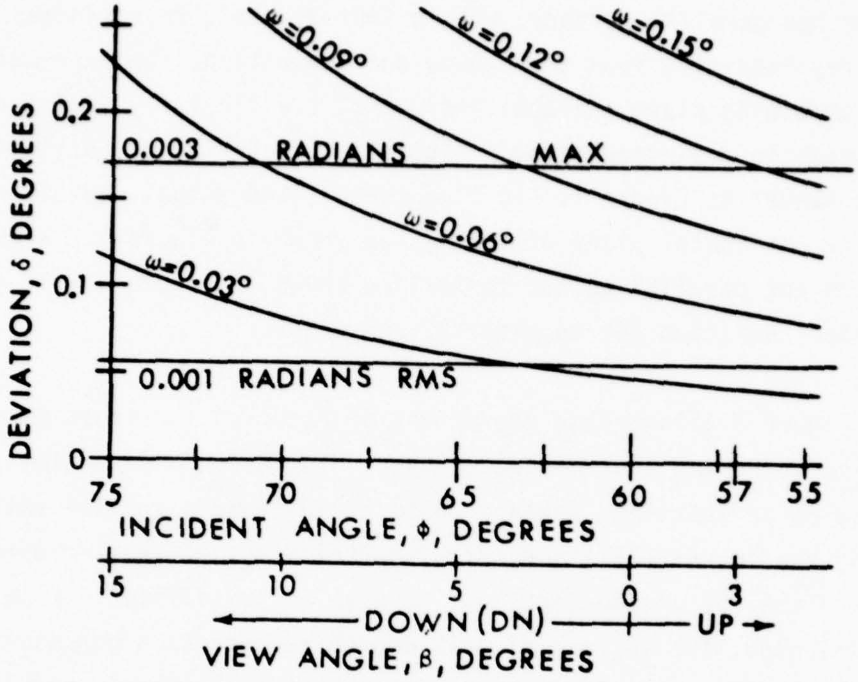


Figure 2. Optical Deviation - Binocular, Pilot's Display and Target.

a total for the canopy and combining plate. Although the combining plate has parallel surfaces within approximately four minutes of arc, the ray emanating from the canopy surface will not be perpendicular to the combining plate surface; therefore, the final ray will show an increase in deviation error. Figure 2 illustrates the deviations in the same manner as Figure 1. In like manner, the actual deviations of rays in the horizontal plane will occur as shown in Figure 1. Binocular ray traces not parallel to the centerline shown in Figure 2 will produce greater deviation due to material curvature.

Figure 3 illustrates the effect of reducing the cross sectional thickness during the forming process. The graph and section shown are based on an installed surface angle of 30 degrees and the inside surface being the non-parallel surface. The thickness of the laminate reduces toward the top of the section. To give a physical definition of the effect of thinning, the angle ω of 0.12 degree represents a decrease of approximately 0.06 inch in 30 inches starting at approximately the 15-degree down vision angle and ending at approximately the 3-degree up sight line. The 0.003 radian maximum and the 0.001 radian RMS represent the angular deviation limits for the F-16 A/B.

In summary, Figures 1, 2 and 3 were presented as a definition of a problem that exists when curved transparencies, the gun-sight display, the related avionics of the fire control system, and the pilot form an integrated system of the air vehicle. The establishment of a set of design and fabrication requirements must include the method of testing before a set of requirements can be quoted with some confidence of success in their being met. A method for part of the testing procedure is the photographic method outlined in the Distortion Section of this report and Reference 1. This method would provide a photographic recording of the actual grid refracted through the canopy for incremental angular steps between $\pm$ 6 degrees in azimuth and 3 degrees up to 15 degrees down for each of the eyes and the nominal eye position. The basic geometry of the canopy, camera lens, and photographic print are defined in Figure 4.



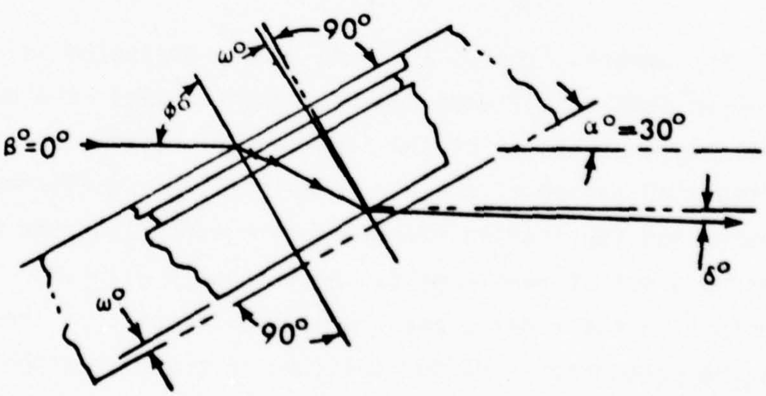


Figure 3. Thickness Reduction Effects - Inner Surface Deviation.

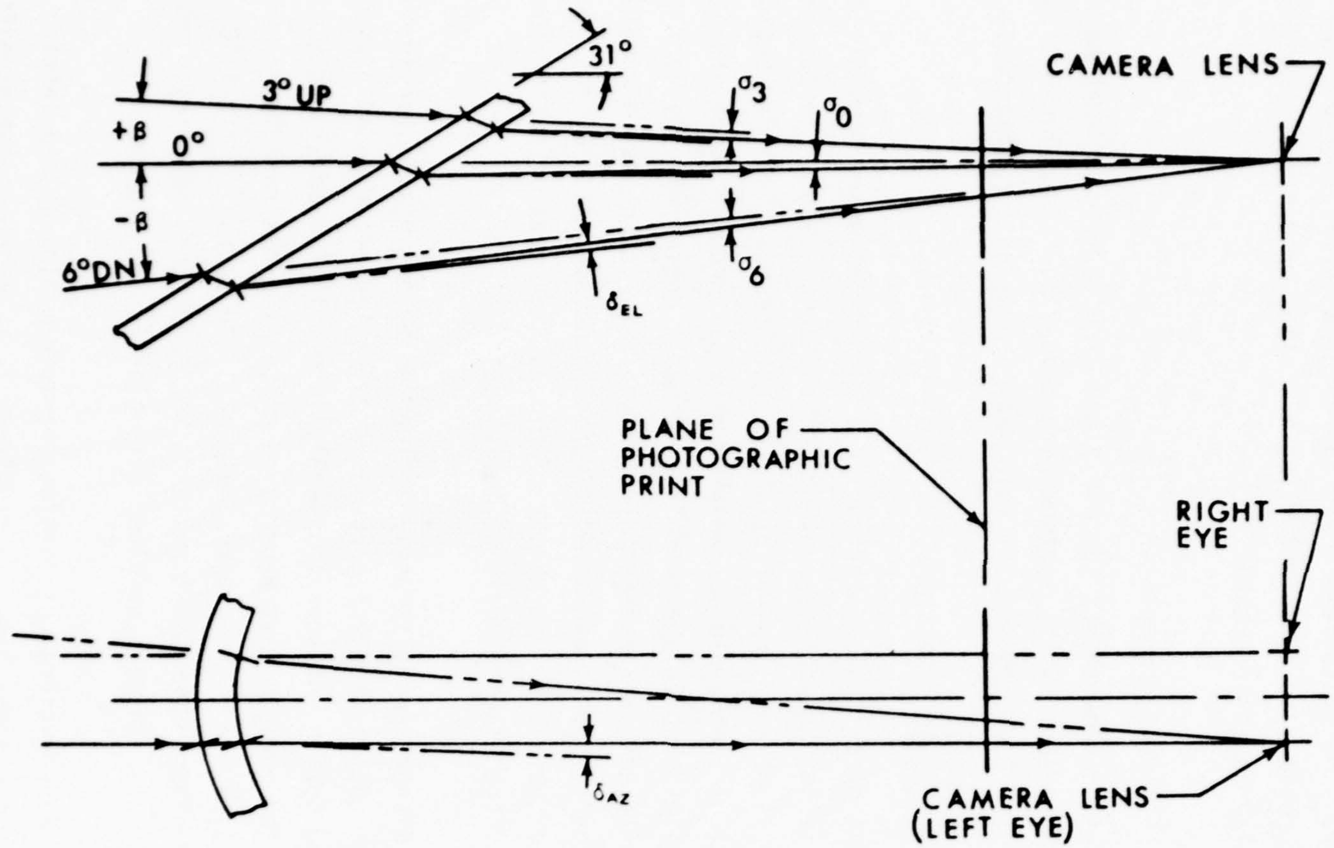


Figure 4. Test Geometry - Canopy/Photography/Camera Lens.

The data obtained from the photographic prints would represent the true relationship of the pilot's sight lines to the true angles external to the canopy. The photograph would represent a vertical grid plane of data that could be used to establish a similar grid plane into the head-up-display computer program.

Angular Deviation Requirements

The deviation requirement for a maximum of 0.003 radian must be maintained since this is the maximum recommended for binocular vision deviations. The 0.001 radian RMS value may or may not be attainable.

Distortion Discussion

The amount of optical distortion in a curved laminated windshield is a function of the variance of index of refraction, angle of incidence, concentricity of surfaces, and variations in thickness. Local distortion will cause a straight line to appear wavy or will distort the shape of objects. Curved windshields which are installed at small angles with respect to the horizontal will have an inherent distortion. In such cases, a straight line will ideally appear as a smooth curve, generated as a direct function of windshield curvature, the number of plies, the index of refraction of each ply, and the viewing angle of incidence.

Optical distortion is strongly influenced by the manufacturing processes and quality control procedures during fabrication of each laminate, and during subsequent assembly into a laminated windshield. A survey of the various test reports on distortion has revealed a spread of the grid-line slope from approximately 1:3 to approximately 1:10. These values depend upon the curvature, the number of laminates, and the use of either all plastics or glass with plastic interlayers.

There is an additional distortion characteristic, called lensing, that requires consideration as a requirement. Lensing is the magnification of visual images as the result of local variation in thickness between two or more surfaces of a transparent panel.

Lensing, unless very severe, may not be readily discernable during normal flying conditions, since the eyes are focused on infinity and the parallactic angle is less than one minute of arc. At approximately 360 feet, the parallactic angle is approximately two minutes of arc, and at approximately 72 feet the parallactic angle is ten minutes of arc. This increase in the parallactic angle is the convergence of the two eyes on an image and under normal conditions this does not produce any ill effects.

Lensing may produce eye fatigue, however, if the eyes are forced to diverge and converge with respect to each other when viewing through a transparent panel. Under normal conditions, the parallactic angles stated above are twice the amount each eye moves. As noted in Reference 2, ten minutes of arc is considered to be the maximum tolerance for divergence, convergence, and subvergence of the eyes in adjusting for binocular deviations. If it is assumed that one eye remains stationary and focused at infinity, five minutes of arc would result in the convergence of the eyes on an apparent image at approximately 144 feet; ten minutes of arc would result in the convergence of the eyes on an apparent image at approximately 72 feet. This discussion gives relative values to the effects that deviation and distortion of a transparent panel would have on an observer's eyes.

Distortion Requirements

The recommended maximum values for grid-line slope to limit optical distortion due to angular deviation are as follows:

> Maximum grid-line slope shall be 1:15, when the grid is photographed through the windshield in the installed position.

Scratch and Inclusion Requirements

The selection of the recommended scratch and inclusion limits shown in Table 1 was based on current criteria utilized by the aerospace industry. Further expansion of these limits should be based on tests.

These requirements imply testing by visual means only; therefore, some defects may not be detected easily by the human eye. It is recommended that edge-lighting and polarized lighting be used as aids to the visual inspection technique,in order to detect internal stresses and potential "rainbow" defects.

Double Image Discussion

Double or multiple images will occur normally at night, as the result of reflections from each surface of a transparency. This will occur in flat panels only when the wedge angle is large enough to reflect the images back to the observer's eye. For curved mutli-laminated windshields, this effect can occur at high angles of incidence; for example, from runway lights seen in the lower portion of the windshield. The early detection of this problem, during the design stages, is very difficult. If found during later stages of design, however, changes in geometry may no longer be feasible. The use of one "thick" ply within a laminated windshield will tend to reduce the number of secondary images, because of the longer reflected ray path-length and the absorption characteristics of the material.

Double Image Requirements

No further recommendations for requirements in this imaging area are considered at this time.

Reflected Image Discussion

Reflected images generally occur as a result of the internal light being reflected into the observer's eye. This can be a very serious disturbance to pilot vision during night flying.

TABLE 1. OPTICAL DEFECT ALLOWANCES

TYPE DEFECT	MAXIMUM SIZE ALLOWABLE	MAXIMUM NUMBER PER PANEL
1. Scratches (1)	0.020 inch width, 0.005 inch depth or 3 inches in length	Items 1 - 3 Each panel shall be individually evaluated. When the size number, and/or grouping of these type defects cause a distraction to vision or vision impairment, the panel shall be rejected.
2. Lint or Hair (3)	3 inches in length	
3. Smears and Rubs (2)	5/8 inch width or 1-1/2 inch length	
4. Translucent Inclusion or Imbedded Particles (4)	0.125 square inch in area (3)	Items 4 & 5 The total number of these type defects between 0.035 and 0.125 square inch in area for translucent defects or between 0.035 and 0.070 square inch in area for opaque defects shall not exceed twelve (12) per panel. Defects up to 0.035 square inch in area shall be acceptable provided they do not cause vision impairment or visual distraction.
5. Opaque Imbedded Particles and Inclusions (4)	0.070 square inch in area (3)	
6. Surface Crazing	None allowed	

NOTES: (1.) Defect Number 1 allowed only on acrylic surface.

(2.) Defect Number 3 allowed on acrylic surface or polycarbonate surface prior to coating.

(3.) Defect Numbers 2, 4 and 5 allowed only within interlayer material.

(4.) The selection of polycarbonate materials should be accomplished to minimize potential pilot objections to inclusions affecting visibility and whenever possible the elimination of inclusions in the areas where maximum stretching will occur during forming.

Reflected Image Requirements

The use of anti-reflection coatings could reduce the amount of reflection; however, their use in the environment which is imposed on an aircraft windshield is not considered to be feasible at this time, due to the limited durability of the available coatings.

Realistically, the control of image reflection should be imposed on all internal lighting, internal surface coloring, and even on the wearing apparel of the flight crew.

Binocular Vision Discussion

Normal eyes are capable of both convergence and divergence in the horizontal plane and supervergence in the vertical plane, in order to adjust for small amounts of binocular deviation. The most difficult adjustment is in the vertical plane. This deviation may also be referred to as a difference between (1) the true parallactic angle subtended at the object by the base distance between the eyes, and (2) the actual angle of the rays passing through a transparent panel. The eye has its least tolerance for adjustments in the vertical plane, and a limit of 10 minutes of arc has been defined as the maximum allowable.

Curved windshields will produce a geometrical deviation of the parallactic angle in both the horizontal and vertical directions. The amount of curvature, the installed angle, the distortion and the angular deviation of the windshield will all affect the parallactic angle. The continual adjustment by the eyes for parallactic angle errors will lead to eye fatigue and to possible errors of depth perception.

Parallactic angle error limitations should be stated as a design requirement with respect to a defined critical windshield area; thus representing a further evaluation of the effects of distortion and related characteristics which affect the observer's eyes and visibility. Quantification of the allowable parallactic error will depend upon the

most critical usage of each windshield; such as for landing, refueling, gun-sighting and/or HUD (Head-Up Display) requirement.

Two methods should be considered for determining the parallactic error: (1) Analytically determine the geometrical effects by the use of a computer program for images originating at infinity, for runways at different ranges, and for the refueling receptacle; or (2) Conduct a test using a binocular camera to photograph a standard grid-board to determine the distortion or the procedure discussed previously under "Distortion".

Resolution Discussion

Resolution is defined as the ability to distinguish clearly between two objects that subtend very small angles to the eye. The actual resolution requirements will depend upon the viewing purpose of a transparency. For example, a pilot's windshield versus an observer's window represent very different visual task applications. The resolving capability of the human eye, when viewing through a transparency, affects its target-detection capabilities for both high and low target-to-background brightness contrast ratios.

A method for testing the resolution of a transparency, described in Reference 2, would involve the use of a collimator and a telescope. The principle of this method is based on the generally accepted practical limit of resolution for the eye; to wit, one minute of arc under well defined and acceptable conditions of illumination. The test is performed with the laminate placed in its installed position. The telescope is placed at the observer's eye position and received a projected image of a "double-cross" from the collimator. The lines of the cross are spaced to subtend an angle of two minutes of arc, and are one minute of arc wide; therefore, the space between the lines will be one minute of arc. Any deterioration of the cross, or fading, represents a loss in image resolution. This

principle is very similar to a procedure which is used for optically checking camera lenses. An additional feature adopted for camera-lens testing is the use of high, medium and low-contrast targets. The procedure described above for windshield testing could be modified to effectively use both high and low-contrast targets. The density ratio of the lines to their background would be established to represent target/background ratios which may be encountered during flight. One additional modification recommended for this procedure is the addition of a photographic recording capability. This would provide a permanent test record, and would also eliminate judgment-type decisions.

ENVIRONMENTAL DESIGN

The major technical area of environmental design covered in this section includes the topics of thermal design requirements, rain removal, de-fog protection and materials compatibility. There is no anti-icing requirements on the F-16; therefore, that system was not considered.

Thermal

Thermal Technical Discussion

The canopy must survive all thermal conditions to which the aircraft is exposed and must suffer no deleterious consequent effects. This includes both the cold atmosphere on the ground and those at high-altitude, low-speed cruise. It must also withstand both the hot atmosphere conditions during high-speed, low-altitude maneuvers, and those encountered during supersonic dash operations.

Due to the low thermal conductivity in combination with the high thermal mass of the canopy, high thermal stresses are set up by the large temperature differentials within the canopy. It is important, therefore, to know the transient states of the aircraft fairly accurately, such as descent and climb.

During supersonic dash, some canopy materials may exceed their temperature capabilities. At elevated temperatures, these materials

may bubble, change in elasticity, or otherwise change their optical qualities. There is no thermal nuclear requirement on the F-16; therefore, it will not be included in the analysis.

Thermal Requirements

There are two requirements which affect canopy design from a thermal standpoint. One requirement is that the inboard surface of the canopy shall not exceed 160°F. This requirement is given in MIL-STD-38453A. The other requirement is that the materials used in the construction of the canopy be able to withstand the temperature environment experienced by the F-16. The temperature environment of the F-16 is defined as -80°F external surface temperature for the cold condition. The hot condition consists of a soak at 160°F followed by a ten (10) minute exposure to 273°F over most of the canopy and 290°F over the forward portion. This requirement is noted in Reference 3. The above noted conditions are based on the performance requirements for the aircraft. Hot and cold atmosphere are those defined by the worldwide extremes in MIL-STD-210B.

Bird Impact Thermal Discussions

It is expected that over 90 percent of all bird encounters with aircraft will occur below 8000 feet AGL (above ground level). (REF 13)

Temperatures were derived from analysis for the following conditions:

1. Per Reference 3, the minimum external surface temperature for the F-16 is -80°F. This was assumed to occur during high altitude subsonic cruise. To compute the minimum temperature for use during bird impact tests, it was assumed that the F-16 made a 100-second descent to 8000 feet in a MIL-STD-210B cold atmosphere just prior to bird impact. This resulted in an average canopy temperature of -35°F for the half-inch monolithic polycarbonate canopy.

2. Likewise, Reference 3 defines the maximum temperature as a ten (10) minute exposure to 290°F on the forward part of the canopy. The maximum birdstrike temperature would occur following a 100-second descent to 8000 feet in a MIL-STD-210B hot atmosphere. This resulted in an average canopy temperature of 195°F for the half-inch monolithic polycarbonate canopy.

Bird Impact Thermal Requirements

Utilizing the criteria established in MIL-STD-210B and an interpretation of the F-16 operational performance requirements below 8000 feet, noted in Reference 3, the temperature requirements for a series of potential F-16 canopies are noted in Table 2.

<u>Rain Removal</u>

Rain Removal Technical Discussions

Based on the fact that adequate visibility for maneuvering the aircraft must be maintained during heavy rain, a rain removal system must be provided.

There are two alternatives. One is a ground applied rain repellent which is usually applied prior to each flight. The other is a system installed onboard and is applied after the canopy is wet. The latter system requires some means to spread it over the canopy surface. This could be accomplished by a wiper or a jet blast. A wiper system can not be used on the F-16 because of the compound curvature of the canopy.

The onboard system is probably not a viable system for the F-16 because of space limitations and no jet blast supply source in the area of the forward canopy.

The ground applied rain repellent fluid will need to be compatible with the outer surface material used on the canopy. This will then dictate the type of fluid used. The ground applied fluid is acceptable

TABLE 2. EXPECTED TEMPERATURES AT BIRDSTRIKE

CANOPY MATERIAL	AVERAGE BIRDSTRIKE TEMPS (°F)	
	MAXIMUM	MINIMUM
1/2-In. Monolithic Polycarbonate	195	-35
5/8-In. Monolithic Polycarbonate	190	-30
3/4-In. Monolithic Polycarbonate	185	-30
7/8-In. Monolithic Polycarbonate	185	-30
1/2-In. Polycarbonate 0.08-In. Acrylic CIP Interlayer	190	-30
5/8-In. Polycarbonate 0.08-In. Acrylic CIP Interlayer	185	-30
3/4-In. Polycarbonate 0.08-In. Acrylic CIP Interlayer	180	-25
7/8-In. Polycarbonate 0.08-In. Acrylic CIP Interlayer	180	-25

for the F-16 due to the relatively easy accessibility to the forward canopy from the ground.

Rain Removal Requirements

The following requirements were identified in Reference 3.

1. The F-16 canopy shall withstand the worldwide ground precipitation extremes described in MIL-STD-210B.

2. The F-16 canopy shall withstand the operational inflight rainfall rate of 0.59 inch per hour, specified in MIL-E-38453A.

Defog System

Defog System Technical Discussion

To maintain adequate visibility for the pilot during all phases of flight, a defog system for the internal surface of the canopy must be provided. The defog system shall maintain the canopy fog-free during exposure to the transient and steady state atmospheres defined in MIL-T-5842A.

There are two systems normally utilized to meet the defog requirements. These are an electrical system and a hot air defog system. The electrical system uses a type of heater which is an integral part of the canopy.

The other system is a hot air system which takes a warm air supply and directs the air across the internal surface of the canopy. This system has some disadvantages in that the system is usually noisy and the air causes discomfort to the pilot. If the pilot is in a flight suit with helmet and visor, neither of the two disadvantages would be important.

Defog System Requirements

The following are the requirements for the defog system for the F-16:

1. The defog system shall meet the intent of MIL-T-5842A over the forward portion of the canopy.

2. The fogging environment to be considered shall be as defined by MIL-T-5842A.

3. A means shall be provided to defog the inner surface of the forward portion of the canopy.

Materials

Materials Compatibility Requirements

The materials selected for development into an F-16 canopy transparency, after having undergone the various fabrication and forming processes, shall be capable of performance in accordance with the environments expected in operation but shall also withstand any combination of environments as follows:

1. The transparency shall withstand storage temperatures within the range of -65°F and 200°F.

2. The transparency shall withstand the effects of relative humidities up to 100 percent at temperatures up to 150°F without cracking, clouding, increase in haze, loss of light transmittance abrasion resistance, or adhesion.

3. The transparency shall be capable of withstanding exposure to fungus growth as encountered in tropical climates. All fungus nutrient materials other than those used in hermetically sealed assemblies shall be treated in such a manner as to prevent fungus growth.

4. The transparency shall be abrasion resistant or so processed as to withstand salt spray when the transparency is in the "as installed" condition.

5. The exposed surfaces shall be capable of withstanding exposure to solutions commonly used in the operational and maintenance environments without degradation. The following list of solutions shall be used and shall not be considered as a restrictive list.

   a. Mild soap and water
   b. Airplane wash cleaner fluid (MIL-D-8514)
   c. Phosphoric acid cleaner (A-3) (MIL-C-5410)
   d. Alodine 1200 spray and rinse (MIL-C-5541)
   e. Bug remover fluid (P6009)
   f. Rain repellant fluid (Repcon. FSN 6850-13905297)
   g. Naptha
   h. Jet fuel (JP-4)
   i. Isopropyl alcohol
   j. De-ice fluid (ethylene glycol type).

6. The surfaces shall withstand sustained abrasion from normal cleaning as well as rain impingement up to 420 knots at sea level. The 420-knot requirement is from the flight regime selected for the European environment.

ELECTRICAL/ELECTROMAGNETIC DESIGN

An aircraft can be expected to encounter one or more of several hostile electromagnetic (EM) environments. The EM environments of possible concern are; (1) triboelectric (p-static) charging, (2) lightning phenomena, (3) external high power radar or similar radiation, and (4) electromagnetic pulse (EMP) radiation.

While it is possible for all of these EM environments to be present, the relative importance of each to the tactical operation of the aircraft is determined by factors which are not the direct objective of this canopy design investigation.

Other non-electromagnetic environments and constraints also influence the optimum approach one should take to accommodate the EM environments. Examples are the icing environment and the visual transmission requirements.

Design provisions to accommodate each of the EM environments are not totally independent and an optimum design will require a coordinated approach. The following paragraphs discuss each of the EM environmental influences on a canopy design. The initial assumption is that all four of the EM environments will be present to influence the design. However, design simplifications will be discussed when the man-made radar and EMP environments are not of tactical concern.

Triboelectric (P-Static)

Triboelectric (P-Static) Phenomena Technical Discussion

Triboelectric charging of the aircraft occurs primarily from the impact and subsequent rubbing action of ice crystals, snow, raindrops, and dust particles. When the impact is on a non-conducting surface, such as an aircraft canopy or windshield, the charge is not free to migrate, as would be the case for impact on a metal surface, and will build in intensity until the charge potential exceeds the insulating qualities of the canopy surface or the air adjacent to it. The result can be a transitory electrical discharge which generates electromagnetic radiation and visible, sometimes spectacular, sparking.

If the canopy design employs electrically conductive coatings or wires beneath the outer surface, large electrical transients can be induced into these conductors. If these conductors are electrically connected to some electrical device, such as a canopy anti-icing or defogging heater control, severe damage might result to the control as a direct result of the electrical transient. If the conductive surface is passive and is not electrically connected to anything, as might be the case for solar or radar reflective coatings, the voltage in the conductive

coating resulting from the surface discharge transient can arc to nearby aircraft metal structures, usually through the plane of the transparency.

Triboelectric action can deposit a charge on a canopy regardless of whether or not there is a conductive coating or conductors buried under the outer surface. However, the consequences of the discharge can be far less severe if there are no conductors under the outer surface. When conductors are present, the canopy material becomes the dielectric of a capacitor formed by the conductor on one side and the bound surface charge on the other side. Energy is stored in this capacitor and the dielectric is stressed by the voltage differential between the outer and inner surfaces of the material. Therefore, the possibility of electrical puncture of the canopy material as the result of triboelectric charging is much higher when electrical conductors are present under the outer surface than when no conductors are present. Discrete wires also present a higher dielectric stress than does a uniform conductive coating because of the higher local electrical gradient in the vicinity of a wire.

Triboelectric charging is a natural phenomenon which will occur on any airplane; therefore, an adequate design must account for the charging process. The ideal approach is to eliminate the bound charges by providing conductive surfaces over all of the aircraft external surfaces including the canopy. For the canopy, this can be effectively accomplished by providing a transparent electrically conductive surface coating for the outer surface of the canopy, and by adequately connecting this coating to the metal skin of the aircraft. Unfortunately, permanent, transparent conductive coatings are difficult to achieve. The present state-of-the-art only provides these coatings for glass material, and the degree of permanence is not well established. No permanent or semi-permanent anti-static coatings are known to be available for satisfactory application to aircraft windshields and canopies which have plastic outer surfaces. Some pre-flight wipe-on compounds have been developed which appear to control static charging, but they generally last for only one flight or less.

For non-electrical reasons, the present canopy studies are oriented toward plastic outer surfaces, and the candidate materials have good dielectric properties. This means that the surface and bulk resistivity of the material cannot be counted on to self-discharge at a rate higher than the possible triboelectric charging rate. Therefore, some other discharge method must be applied, or the effects of large surface charges and discharges must be accommodated in the total aircraft design.

Over the years various methods other than anti-static surface coatings have been investigated which might prevent detrimental charge buildup on the surface of the canopy. These included neutralizing the charge with a locally generated counter charge, liquid sprays, etc. None appears electrically too practicable except the following described method. This method employs a series of narrow electrical conductors placed on the canopy surface and grounded to the metal airframe fuselage. The conductors are arranged in an open grid or as fingers to reduce the total canopy frontal surface to a number of smaller areas. This technique reduces the effective charging area and potential to which the charge can build before it discharges to a metal conductor. This approach can be fairly effective electrically, but it has some significant disadvantages in a practicable installation. For example: the surface conductors may be a visual distraction which is always present; the strips and their electrical connection to the airframe may present an additional maintenance problem; and the design and application are a small but real additional cost. The presence of external conductors on the canopy surface will also influence the attachment of lightning, both for the direct and swept stroke cases. The lightning impact may not be harmful, but its influence would require thorough investigation and test.

A more detailed discussion of static charge control is contained in Section V of Reference 4.

Triboelectric (P-Static) Requirements

No military specifications have been found that apply specifically to aircraft windshields or canopies; however, MIL-E-6051D "Electromagnetic Compatibility Requirements - System", Paragraph 3.2.10.2, Conductive Coatings, states the following:

> "Conductive coatings shall be applied to all non-metallic materials on the exterior surfaces of airborne vehicles that are exposed to airflow. After application, coating resistance shall measure not less than 10 megohms and not more than 50 megohms per unit area at any given point".

Triboelectric (P-Static) Control Design & Recommendations

A design recommendation for the control of p-static discharges should not be made independent of the influence of other EM environments, and aircraft design constraints. For reasons which will be discussed later, the influencing EM environments are lightning and radar reflectance. Radar reflectance [radar cross section (RCS) control] will place a highly conductive coating under the outer canopy surface. This increases the probability of surface discharging and influences the required thickness of the outer surface material to prevent puncture due to static electric charging.

The previous paragraphs have discussed the static charging process and potential discharge methods. Surface mounted conductors appear to be the most practicable method of controlling the charge buildup. However, this technique does not appear consistent with the design objectives of a low cost, low maintenance aircraft and is not recommended for the present. Fortunately, the conductors can be added at a later time if warranted by flight experience.

Since electrical testing is not a part of this design study, experience from other related programs must provide the design guidance. The static electric and swept stroke lightning tests of References 4 and 5 and the canopy lightning tests on the F-15 aircraft canopy design of Reference 6 lead to a qualified conclusion that an outer layer of 0.080 thick acrylic would probably prevent dielectric puncture in the presence of static electric charging. In any event, high voltage testing of a prototype part is recommended.

A peripheral external surface bus is recommended to provide control for the grounding of surface discharges, even though an anti-static surface coating is not practicable. This technique is discussed in further detail in Section V of Reference 4. The peripheral bus and associated metal static drain strap must be installed with the expectation that at some time they will be called upon to provide a low impedance lightning path to the fixed portion of the aircraft fuselage. This will be discussed in more detail under the sections on lightning.

Aircraft antennas, particularly low frequency receiving antennas should not be located in close proximity to the frontal area of the canopy, when this is possible. This recommendation is made to reduce the coupling of EMI to the antenna from radiation from canopy surface discharges. It is recognized that factors other than susceptibility to p-static may have higher priority on antenna locations. Therefore, the possibility of p-static coupling should be considered when determining antenna locations.

Static Electricity Safety Considerations

When the outer surface of a windshield or canopy cannot be coated with an effective anti-static coating, the surface can be expected to accumulate an electrostatic charge. This charge could become an electric shock hazard when either the inside or outside surface of the transparency is touched.

When no grounded electrically conductive coatings are employed within the transparency to provide an electrostatic shield, contact with the inside surface of an externally charged surface will cause an immediate redistrubiton of the external charge. The result is an induced charge in that part of the body that comes to or touches the inner surface. A potentially severe shock can result.

Close proximity to or touching the outer surface of a charged transparency can provide a discharge path and result in an electric shock. The shock may not be fatal in itself, but it can cause surprise or uncontrolled muscular reactions which may cause one to fall, with possible serious consequences.

The length of time a charge can be retained depends on the initial charge, the time between receiving the charge, and the insulating qualities of the outer surface of the transparency. Laboratory tests (References 4 and 5) have demonstrated the ability of acrylic transparency material to retain a substantial electrostatic charge for over 18 hours after the removal of the charging source. In contrast, higher loss soda-lime glass could retain a charge for only seconds to a very few minutes.

From these experiments it may be concluded that an acrylic surfaced windshield or canopy could easily retain a dangerous charge even after the time lag between the inflight encounter with ice crystals and the landing and parking procedure. This conclusion is especially valid when the intervening weather and runway conditions are dry. Currently, a soda-lime glass windshield presents a minimum shock hazard. Ground operations procedures should provide for discharging the transparency or for very careful handling of a canopy.

Lightning

Lightning Strike Technical Discussion

The electromechanical configuration of the canopy area of the F-16 is such that it is possible for this area to be a candidate for a Zone 1 direct lightning strike. This same situation has applied to other aircraft and extensive testing has been conducted on some aircraft. The F-16 design team has studied this prior work, and most specifically the F-15 studies. The F-15 final report (Reference 6) concludes that lightning strikes to the canopy would preferentially flash over the outer canopy surface to the aircraft structure or to the conductive metal splice plates of the canopy. Further, lightning punch-through of the canopy is an unlikely occurrence, and the addition of a lightning protection device is unnecessary.

The F-15 tests were conducted on polycarbonate material. A mock-up canopy employing 1/8-inch thick polycarbonate material and an actual F-15 canopy were both tested. No punch-through was encountered for the unprotected canopies as well as for the canopies equipped with various internal and external lightning diverter protection methods. A sub-outer surface electrically conductive coating for radar cross section control was not included in the F-15 tests.

The F-15 test program also investigated internal electrical discharge streamering from the pilot's seat and helmet. The intensity of the streamering was compared with the spark intensity from a known low energy source. The concluded implication was that the internal streamering was lower in total energy than the arc from a static spark commonly encountered on a dry day when walking on a nylon carpet.

The above discussion relates to the direct attachment of the initial flash to the canopy area. The canopy may also be exposed to the action of swept stroke lightning (Reference 4). Since the action of swept stroke lightning is expected to be less severe than a direct strike, the F-15 tests should represent a worse case than swept stroke lightning.

When an electrically conductive coating, such as used to control the radar reflection from the canopy area - referred to as radar cross section (RCS) control - is used to cover the whole canopy area, the interaction of the canopy with the lightning will be modified from that of a clear canopy. The F-15 tests lead one to expect that a full conductive coating would not stress the insulation of the canopy more than occurred when the internal diverter strips were employed during the tests. The conductive coating would be expected to produce a less concentrated field condition and, therefore, a lower dielectric stress on the canopy material. However, this is an inferred conclusion, not a conclusion based on the actual test. Other factors, discussed later, might possibly cause other serious problems.

The presence of a total-coverage RCS conductive coating, if properly grounded to the airframe, should alter the pre-strike electric fields under the canopy. Since electric fields are very easily and effectively shielded, one might expect no internal streamering of the type noted during the F-15 tests. This electric field shielding should be maintained during the actual lightning flash to the canopy, assuming the RCS coating remained an active shield during this time.

An RCS coating is not expected to be an effective magnetic shield to the fields of the lightning flash even though the canopy may not be punctured by the lightning action (Reference 4, Appendix B). The intense magnetic field of a lightning flash along the outer surface of the canopy to the fuselage can be expected to induce large currents in any nearby metal conductors that may form a closed conductive loop. These currents can cause localized sparking and even molten metal droppings when resistive contacts form the closed loop path. Shock hazards, not necessarily lethal, may also be present within the cockpit due to voltage drops along conductors carrying the direct or induced lightning current. Direct field excitation of the pilot's body appears possible and might result in an electric shock sensation.

Specific documentation of these adverse magnetic field phenomena are not available for aircraft lightning incidences, but they have been encountered by Douglas personnel when working with laboratory generated simulated lightning. The translation of these laboratory experiences to an aircraft environment appears reasonable.

Other aspects of high current lightning involve the high pressure shock wave that accompanies the discharge, and the mechanical effects that accompany very high current, fast rise time lightning current paths. Work on this and related programs indicates that these two lightning effect areas may have been overlooked during past canopy lightning investigations.

The complex mechanical interface between the fuselage and the canopy of a fighter aircraft may include design features which might inhibit the safe passage of high current lightning. The canopy, or large portions of the canopy, must be capable of separating from the fuselage both for normal ingress and egress, and for emergency egress. Furthermore, the canopy-fuselage interface frequently forms a pressure seal to retain cockpit environmental air pressure during flight.

The pressure sealing and mechanical latch system reduce the low resistance conducting paths between the canopy metal frame and the fuselage to but a few finite points. These mechanical restrictions produce a highly indirect electrical path for the lightning current. These path restrictions force the lightning current to seek a more direct path with attendant large mechanical forces being exerted on the structural members of the path. In practice, the discharge may divide between both types of paths. The result can be a large shock wave from the arc and large mechanical forces developed by the multiple-direction magnetic field interactions. Either or both might be highly detrimental to the mechanical and pressure integrity of the canopy-fuselage interface.

Still another area of potentially serious concern is the effect magnetic pulse fields and arc-created shock waves may have on the detonating systems frequently used for canopy emergency ejection. It is strongly recommended that the possible effects of lightning on the safety of these ejection systems be thoroughly investigated before a specific design is flown.

If initial stroke of swept stroke lightning were to flash across an aircraft canopy without puncturing the canopy the localized current density in a buried RCS coating would be expected to reach potentially high destructive levels. Significantly higher currents could be expected in a direct strike. Because of the resistance of an RCS coating, localized heating and possible destruction of the RCS coating may occur. Additional information on this subject may be found in References 4 and 5.

From the above discussion, it is evident that it would be highly desirable to totally divert any lightning strikes from the canopy area. This does not appear feasible for the aircraft as now configured. The magnetic field exposure of canopy equipped aircraft appears to be an occupational risk that must be accepted, although some reduction of the results of exposure appears reasonable. Fortunately, the probability of a lightning attachment to an aircraft canopy is relatively low - but possible. Specific cockpit designs should stress low exposure of the crew and systems to the effects of magnetic induction fields.

Lightning Requirements

The most specific requirement for lightning protection is found in Specification MIL-B-5087B, Amendment 2, 31 August 1970, "Bonding, Electrical, and Lightning Protection for Aerospace Systems".

"Paragraphs 3.3.4 - Lightning protection shall be provided for all possible points of lightning entry into aircraft and shall be proven by test. The entry points include but are not limited to the following:

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h. Canopies

No specific canopy lightning tests have been made or are planned for the F-16 by General Dynamics. Instead, extensive testing by McDonnell Douglas on the F-15 has been used by General Dynamics for application to the F-16 because of the similarity of the electromechanical configurations of the two aircraft in the canopy area.

Lightning Design Recommendations

When the tactical use of an aircraft permits, it is recommended that no metallic conductors be placed within, and on or near the inside of a canopy. This precaution and the use of a good dielectric material of adequate thickness should help to reduce the probability of lightning attachment to the canopy, and the possibility of related induced current and voltage effects.

When circumstances dictate the use of conductors, such as an RCS coating on or within the canopy, special precautions are required to reduce the probability of lightning related problems. First, the design and development schedule for the aircraft should provide for thorough investigation and testing to determine the presence, extent, and cure of possible lightning related hazards.

The presence of a conductive coating will increase the probability that a lightning flash to a canopy will hug the surface of the canopy as it seeks a conductive path to the fuselage skin. This close coupling of the flash path to the canopy surface may increase the possibility of the flash path entering the complex electromechanical interface between the canopy and the fuselage rather than arcing directly to the fuselage skin. **The more confined and conforming the flash path, the greater the damage caused by the shock wave and magnetic interaction accompanying the flash than if the flash were free to travel directly to the fuselage.**

It is very difficult to precisely predict the path that lightning might take if it were free to choose a path over a canopy to the fuselage. However, use of diverter strips might be a possibility to enhance the probability that lightning to a canopy area would follow a predetermined path to the fuselage. This path could then be given special design attention to reduce the probability of serious damage. The diverter strip has the effect of attracting the localized attachment of lightning on a dielectric area and then conducting the lightning current in an ionized plasma above the dielectric to a predetermined safe area on the fuselage. The diverter strip method has been successfully used on radomes. Its use on canopies has also been investigated (Reference 6), but not on electrically coated canopies. The presence of the electrical coating might alter the pre-strike electric fields and effect the action of the diverter strip. Further investigation and tests would be required. If diverter strips are not feasible, the entire canopy - fuselage joint area must be designed to permit the direct, safe passage of very high lightning discharge currents. Every effort must be made to provide a direct path from the canopy surface areas. If the canopy-fuselage joint is utilized, the probability of structural damage is greatly increased.

Radar Reflection Control (RCS)

RCS Control Technical Discussion

The technical intent of RCS control is to substantially reduce the radar signal reflected from the aircraft and to reduce or eliminate identifying characteristics of the remaining reflection. The area of an aircraft covered by the canopy presents many surfaces which can produce strong radar reflections. The ideal, but presently impracticable, solution would be to absorb all of the radar energy which reaches the canopy area. An alternate, more practicable solution is to coat the canopy with a highly conductive but transparent coating which will reflect the radar signal rather than allow its two way transmission into and out of the cockpit area, as is the general case for clear canopies. Experiments (Reference 4) have shown that adequate reflection at many radar frequencies can be achieved if the windshield is covered with a conductive coating having a surface resistivity

of as much as 30 ohms per square. A lower resistivity is desirable, but a practicable balance between low resistance and high light transmission places the coating resistivity at around 10 to 15 ohms per square.

The RCS coating should be void-free and continuous down to the edges of the canopy, whenever other constraints are not of overriding importance. Because of the fragile nature of most highly conductive transparent coatings they are not placed on the outer surface of the canopy. They are usually located on an internal surface of a multi-layer canopy so that both the outer and inner surfaces are protected from scratches and abrasion.

The RCS coating will be an effective radar reflector even if it is not grounded to the aircraft metal fuselage because the wavelength of the radar signal is usually very small compared to the linear dimensions of the RCS coating. However, the electrical influence of triboelectric charging and lightning dictate a safer installation if the RCS coating is fully grounded to the aircraft structure. An ungrounded RCS coating could reach a potential of many thousands of volts; electrical arcing to nearby structure would occur.

Grounding of the RCS coating should be accomplished by applying a peripheral bus to the coating surface. The bus should then be multi-point grounded to structure by means of low impedance connections to the aircraft structure. Since these grounds may be required to carry hundreds or perhaps thousands of amperes for a few microseconds during a lightning encounter (Reference 1), the path from RCS bus to structure must have low impedance, otherwise high voltages will be developed across the length of the connection and sparking or pilot shock may occur.

As discussed under the previous sections on triboelectric charging and lightning, the presence of a conductive RCS coating under the outer surface of the canopy can provide a beneficial electrostatic shield for the pilot and cockpit equipment. The RCS coating also presents some problems in that it increases the probability of large triboelectric discharges and lightning attachment to the canopy surface.

The electrical stress on the canopy outer surface material is also increased, thus increasing the possibility of canopy electrical puncture. Induced current in the RCS coating due to lightning is expected to increase the incidence of damage to the RCS coating.

The mechanical adhesion of the RCS coating to the canopy materials may be significantly less than the adhesion between the basic canopy materials (Reference 7). Since these coatings and their applications are considered highly proprietary by some of the manufacturers, the possible effect of adhesion on the overall canopy design must be considered.

Radar Reflection Control (RCS) Requirements

No general requirements have been found which relate to the control of the radar cross section (RCS) of aircraft windows, windshields and canopies. The contribution of the canopy to the total RCS of the aircraft must be determined for each aircraft configuration. A decision as to whether or not the canopy must have RCS control measures applied is not one that can be made on a weapons system basis and involves many important factors in addition to the absolute contribution of the canopy and cockpit to the total RCS of the aircraft. However, for the purpose of this canopy study it is assumed that RCS control could be a design requirement.

RCS Control Design Recommendations

The addition of an RCS coating to a canopy can significantly complicate the design and qualification of the final aircraft installation. Therefore, it is recommended that RCS coatings not be employed where their need is not fully justified.

When an RCS coating is required, the design must provide for the possible increase in electrical stress on the material lying between the outer surface and the RCS coating. The coating should be constructed with a peripheral electrical bus capable of conducting significant current for a few microseconds (References 4 and 5). Bus cross sections used in conventional electrically heated coating should be adequate. The bus should be

multipoint grounded to the aircraft fuselage by low impedance straps. These straps should be two or more inches wide, but need not be more than a few thousandths of an inch thick. The high frequency pulse current is conducted only on the outer surface of the strap, therefore, thicker metal does not contribute to a better conductor.

The construction of a canopy provides a metal frame to which the transparency is joined. The frame in turn joins to the fuselage in an interface that is usually not conducive to providing the required low impedance path to the fuselage structure. The canopy tiedown points probably represent the best starting point for the location of the RCS grounding straps, as the remainder of the interface is usually a non-metallic seal. Because of the electromechanical complexity of most of the grounding concepts studied so far, it is strongly recommended that the final design candidate be tested. The testing should be combined with tests on the canopy surface flash path grounding to be provided for triboelectric discharges and lightning flashes.

Electromagnetic Pulse (EMP) Requirements

EMP requirements, like RCS requirements, are established on the basis of the tactical use of the weapon system. Prior studies, Reference 1, have shown that the primary interaction of a windshield (or canopy) with the EMP environment is with respect to the signal received by canopy electrical coatings and conducted to connected or coupled electronic systems. Conductive transparent coatings do not afford significant electromagnetic shielding for many applications.

STRUCTURAL DESIGN

The major technical area of structural design includes the topics of basic structural design, internal and external loading, and hail impact. Bird impact requirements will be covered separately.

The structural design requirements must be met by applying all applicable design loads to the appropriate area and/or point of the aircraft structure.

The basic structure shall be designed to maintain its structural integrity when subjected to the required load effects. These include all imposed external aerodynamic loadings, internal pressurization, and inertia loading, in addition to hail impact, temperature-induced loading, structural carry-through loads, as applicable, and loadings incidental to the emergency operation of the canopy.

Structural

Structural Technical Requirements

Structural design requirements are derived in part from the Design Service Life and Design Usage based on the usage definition and mission profiles provided by the Air Force Lightweight Fighter Project Office. These and other applicable structural design requirements are presented in General Dynamics Report 16PS007B, (Reference 8), "Structural Design Criteria", 15 March 1976. This report presents the structural design criteria for the F-16 Fighter. It is applicable for the structural design and analysis of the operational aircraft. Subject criteria follows format and content of MIL-A-8860 series military specifications, Revision A. Subject document is applicable to both the F-16A and F-16B aircraft.

Reference 8 also establishes, pursuant to Air Force requirements, the projected usage categories identified as training and combat and further subdivided by mission type, and mission profile information. The projected aircraft usage results in the following design service life information:

Total Flight Hours	=	8,000
Total Number of Flights	=	5,754
Total Number of Landings	=	6,555
Total Service Years	=	15

Douglas selected, for the canopy service life, 2000 hours, which represents 25 percent of the aircraft life. The Air Force, however, has stipulated that the canopy must withstand two lifetimes of pressure and temperature cyclic conditions which would be an equivalent of 4000 hours.

Internal/External Loading

Pressurization Requirements

To properly ascertain the ultimate pressure required for canopy sizing, the correct applied pressures and tolerances must be summed up.

In the information furnished there seems to be a variance in the ultimate pressure (internal plus external). In part, this seems to stem from a possible misapplication of the required internal pressure regulator valve tolerance under limit conditions.

General Dynamics Report 16PR263 (Reference 9) states that the maximum internal pressure for the canopy closed condition is 12.45 psi. This is the design loading on the canopy.

Reference 8 also defines the internal pressure conditions as follows:

Maximum Internal Pressure	= 5.00 psi
Tolerance on Regulator Valve	= 0.10 psi
Net Internal Pressure	= 5.10 psi
Limit Maximum Internal Pressure	= 1.33 x 5.10
	= 6.80 psi
Ultimate Maximum Internal Pressure	= 1.50 x 6.80
	= 10.2 psi

It would appear that the loading used in the General Dynamics mathematical stress analysis model considered the 2P (= 10.2 psi) case only.

In running this program, General Dynamics coupled the canopy model to the forward fuselage model. This provides the appropriate flexible support for the canopy. General Dynamics states that the model output includes hook and hinge loads in addition to stress distribution throughout the canopy structure.

There are, however, conditions which generate a larger total effective canopy internal pressure than the 2P case. General Dynamics report 16PR178 (Reference 10), indicates a total limit internal plus external pressure including referenced port error (+1.0 psi) of 8.8 psi limit. It could not be determined whether the pressure regulator valve tolerance or magnitude was meant to be included. No value was so stated.

Applying the appropriate limit-to-ultimate factor of 1.5 results in

$$P_{ULTIMATE} = 1.5 \times 8.8 \text{ psi}$$
$$= 13.2 \text{ psi } \underline{\text{ultimate}}$$

It would appear that this is the ultimate effective internal pressure at the canopy centerline at FS187.5. It could not be determined from the data furnished, whether this, or a higher pressure, was used in the pressure qualification of the F-16 canopies.

Based on the data available, the number of pressure cycles for one transparency lifetime was defined as 1814 cycles (2000 hours) distributed as follows:

1. Subsonic cruise - standard atmosphere, 916 pressure cycles.
2. Subsonic cruise - hot atmosphere, 102 pressure cycles.
3. Supersonic cruise - Mach 1.6 - standard atmosphere, 498 pressure cycles.
4. Supersonic cruise - Mach 1.6 - hot atmosphere, 55 pressure cycles
5. Supersonic cruise - Mach 2.0 - standard atmosphere, 219 pressure cycles.
6. Supersonic cruise - Mach 2.0 - hot atmosphere - 24 pressure cycles.

The canopy shall be capable of withstanding two lifetimes of testing at these operational pressures and temperatures. One lifetime shall consist of the 1814 cycles. The internal pressure shall be 5.1 psi. The temperatures shall vary between 290°F and -80°F.

The canopy shall sustain a proof pressure of 6.8 psig internal air pressure at room temperature for 15 minutes.

The canopy shall sustain an operational ultimate pressure of 13.2 psig at 200°F.

Hail Impact

Hail Impact Protection Technical Discussion

The Douglas recommended design has an outer ply of "as-cast" acrylic which has a tensile elongation at failure of approximately four percent. The degree of the desired ductility ranges from the brittleness of glass through the intermediate ductility of as-cast acrylic to the preferred high ductility of stretched acrylic. Figure 5 shows the stress-strain relationship of glass, as-cast acrylic, stretched acrylic, and an aluminum alloy. The use of stretched acrylic for the outer ply is limited because of its tendency to revert to its unstretched dimensions when subjected to elevated temperatures.

Hailstone Size Criteria. MIL-STD-210B, dated 15 December 1973, includes criteria for the hail size to be considered. To design the outer panel for a one percent risk would require the panel to withstand the impact of a 2.1 cm. diameter (0.81 inch) hailstone at an altitude of 4 to 6 kilometers (13,100 to 19,700 feet), or a 1.1 cm. diameter (0.42 inch) hailstone at sea level. It is important to recognize that these data are recorded for a worldwide air environment. Ground environmental conditions differ, however. These conditions are based on surface weather observations over land areas.

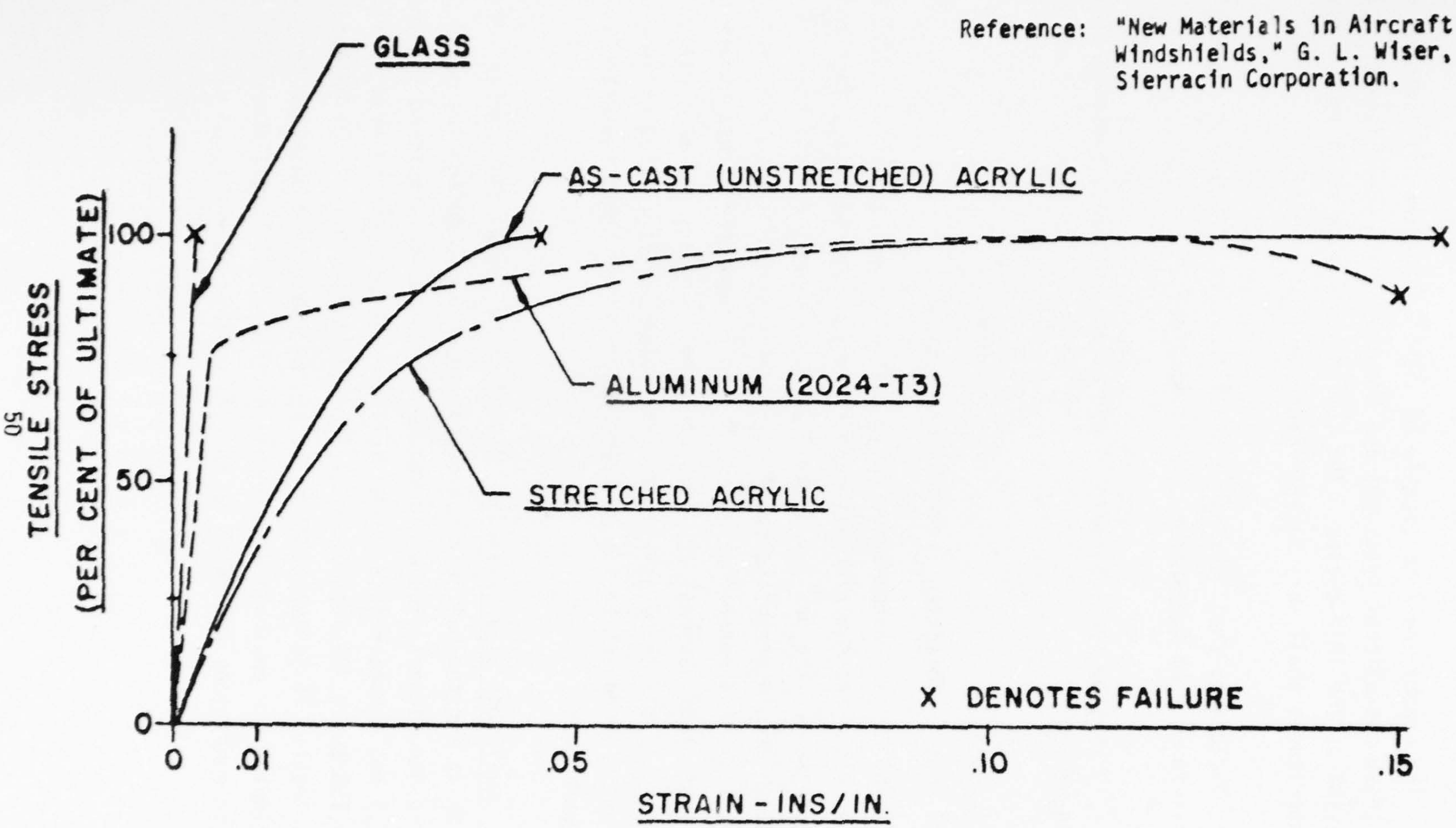

Figure 5. Stress-Strain Curves - Glass, Acrylic and Aluminum at Room Temperature.

In order to satisfy the adopted design requirement for a one percent risk, encounters with a 2 cm. diameter (0.8 inch) hailstone must be considered. In order to design for a 0.10 percent extreme case, a 5 cm. (2.0 inches) hailstone would be considered; however, this is not adopted as a design requirement. A summary of MIL-STD-210B requirements is shown in Table 3.

Risk of Hail Encounter. In Reference 12, an expression was derived for the risk of hail encounter on the assumption that no avoiding action was taken, that the diameter of a hail cell or shower was 1 nautical mile, and that the duration of a hail shower at a single location was 0.1 hour. The expression derived was that the number, N_0, of encounters per flight hour with hail which is x inches or greater in diameter will be:

$$N_0 = 7.26\ (10^{-5}) NVP_x$$

where:

N = *Number of thunderstorms per year on the ground at the geographical area considered*

V = Aircraft speed in knots at the altitude considered

P_x = Probability of occurrence of hail of diameter x inches or greater during the storm at the altitude and geographical area considered. P_x is assumed to be constant up to mid tropopause, and to decrease by one order of magnitude every 10,000 feet above that level.

Then:

$$P_x = P_0\ 10^{-x}$$

The suggested values for N, mid-tropopause height H, and P_1 (the probability of occurrence of hail of 1 inch diameter or greater) are tabulated below:

TABLE 3. HAILSTONE SIZES AND RISKS

OBSERVATION CONDITION	RISK	ALTITUDE	HAILSTONE		MIL-STD-210B REFERENCE
			SIZE	WEIGHT (2)	
Ground Environmental	1%	Ground	2 cm (0.8")	.009 lb	5.1.15 p. 12
	.1%	Ground	5 (2.0)	.139	
	.1%	Ground (1)	14.2 (5.6)	3.05	
World-Wide Air	1%	S.L.	1.1 cm (0.42")	.0013 lb	5.3.12 p. 36
	1%	4 km (13,100')	2.1 (0.81)	.0092	
	1%	6 (19,700')	2.1 (0.81)	.0092	
	.1%	S.L.	3 (1.2)	.031	Table XXVIII p. 68
	.1%	4 (13,100')	6.1 (2.4)	.24	
	.1%	6 (19,700')	6.1 (2.4)	.24	

NOTES;

(1) Largest hailstone ever measured

(2) Hail/ice density = 57.3 lb/ft^3

Region	N.	H.	P_1
UK	15	20,000 ft.	
Europe	25	20,000 ft.	1.25×10^{-3}
Denver	80	25,000 ft.	
Singapore	100	30,000 ft.	2.5×10^{-4}

Subsonic Aircraft Hail Encounter Rate (Example) - Consider an aircraft cruising in Europe at 30,000 feet at 500 knots.

$V = 500$

$N = 25$

$P_1 = 1.25 \times 10^{-4}$

Risk per hour of encountering 1 inch hail = 1.1×10^{-5}

In Reference 12, the problems and frequency of inflight encounters with hail are reviewed and the results are presented for hail impact tests conducted on a variety of test panels and aircraft windshields. The materials tested were as-cast and stretched acrylic, polycarbonate, and glass of various tempers. Tests were made with 3/4 inch, 1 inch and 2 inch diameter hailstones, impacting at speeds up to 2,000 fps. Although these tests were of limited extent, the indication is that, as far as penetration or structural failure are concerned, windshields of subsonic aircraft which are bird-proof will also be hail-proof. For supersonic aircraft, this correlation will probably also apply; however, the specific comparison will depend on the relationship between bird-speed and hail-encounter speed. These speeds will likely fall within similar ranges, particularly when related to the much-greater aircraft speed.

Failure of a ply due to a hailstone impact is a complex process. A dynamic load causes stress waves to propagate through the material "plate", as well as producing an overall structural response. When the impact loading is sufficiently high, the propagation of stress waves through the

structural thickness can cause failure by "spalling". This phenomenon occurs in the same order of time, T, that it takes a stress wave to propagate through the thickness; this time period is approximately:

$$T \cong H\ (\rho/E)^{1/2}$$

where:

H = thickness, inches

ρ = material density, lb sec^2/in.^{4}

E = elastic modulus

This type of failure occurs within microseconds of the initial impact and is sometimes referred to as "early time response".

Hail Impact Protection Technical Requirements

No design and/or test requirements for hailstone impact resistance are made for the F-16 canopy. Considering the risk of bird encounter, which is discussed in the following paragraphs, and the stringent bird impact requirements imposed on the canopy, it is felt that hail impact protection is adequate.

BIRD IMPACT

One of the greatest hazards to an aircraft and crew is created by the aircraft encountering a bird during the flight regime. Historically, up to 30 percent of the bird strikes occur in the area of the crew compartment. When these strikes occur on the crew compartment enclosures, it has been shown that the windshields are destroyed leaving holes that cause high acoustical noise levels, the clear vision areas may be destroyed, occasionally crew members are incapacitated or killed, and aircraft have been destroyed because of bird strikes.

Bird Impact Requirements

The requirements that follow are developed around requirements specified by the Air Force by the contract Statement of Work and knowledge gained from prior programs.

The transparency, when mated to an appropriate canopy structure and fuselage support structure, shall be capable of withstanding, without penetration or causing pilot incapacitation, the impact, at any given location, of a four (4) pound bird at a minimum velocity of Mach 0.53 (350 knots) below 8000 feet (AGL) under standard cold, ambient, and hot temperatures.

Considering the operational requirements of the F-16 aircraft the temperature extremes expected are an average cold temperature of -35°F and an average hot temperature of 195°F.

As a requirement of the contract a determination shall be made to establish changes to the transparency design to meet various bird impact speeds of 400, 430, 450, 500 and 562 knot flight regimes. The required geometry changes have been identified and are noted in Section XI.

Bird Impact Probability Determination

If the F-16 mission is assumed to be the same as the F-4, the data presented in AFFDL-TR-73-103 (Reference 13) can be used to estimate the probability of a bird strike on the F-16 canopy. This document reports F-4 bird strikes (1965-1972) as 9.3 strikes per million flight hours on the windshield/canopy. If the frontal area of the F-16 canopy (427 in.^{2}) is proportioned to the frontal area of the F-4 windshield/canopy (975 in.^{2}), the F-16 could be expected to receive 4.0 strikes per million flight hours on the canopy. Since the life of the F-16 is 8,000 hours, the predicted bird strike rate on the canopy per life time is 0.032.

In essence, this is specifying that out of a fleet of approximately 31 airplanes, at least one airplane will be involved with a bird strike on the canopy.

MAINTENANCE PROVISIONS

Good maintenance practices for any aircraft system must be developed during the initial design phase and must be an on-going effort. A canopy system design must consider the maintenance problems associated with the transparency, latching mechanisms, opening electromechanical subsystems, and handling provisions. The canopy and all the associated mechanical subsystems must be designed for interchangeability and replaceability requirements.

Maintainability

Maintainability Technical Discussion

General Dynamics has identified general maintenance requirements in their document, 16PR022 (Reference 14) dated 1 April 1975 and they appear to be adequate.

Maintainability Technical Requirements

Certain key areas must be clearly delineated and procedures developed for the maintenance of an aircraft canopy, such as the F-16 canopy. Specific areas that require the development of detailed information and instructions are identified as:

Servicing - Mechanism and Electrical Components
Cleaning - Transparency
Repairability of the Canopy
Inspection - Structure and Mechanisms
Handling - Transparency and Canopy
Installation - Transparency
Installation - Canopy
Rigging - Canopy Mechanism

Check-Out - Electrical

Sealing - Pressurization

Sealing - Weather

Sealing - Protective

The Statement-of-Work for the Douglas effort establishes the following maintainability requirements:

"The time goal for a complete windshield panel remove and replace on an operational aircraft is three hours for a two-man team using standard tools".

"Only dry seals shall be used for the transparency/support structure mating surfaces".

SECTION III
ASSESSMENT OF GENERAL DYNAMICS' F-16 TRANSPARENCY/SUPPORT STRUCTURE AND MECHANISMS

There are many contributing factors that influence the design engineer during the design of crew compartment enclosures. Often the shape, weight requirements, and aircraft performance force the designer to use state-of-the-art materials that have not been fully developed. The design of structural members is influenced by an initial projected number of aircraft to be built. When a limited number is required, the design selection is based on minimizing costs. This does not always lead to the best possible design; frequently, improved designs are difficult to implement once the aircraft is in production.

Utilizing the requirements established in Section III and the premises noted above as background information, the Douglas assessment and critique of the F-16 canopy, mechanisms, and supporting structure designs were based on the latest innovative concepts and bird impact technologies.

In order to simplify the presentation of the Douglas assessments, the systems requirements have been segregated into major technical areas for individual review. The major technical areas assessed by this section are as follows:

GENERAL ASSESSMENT

VISION/OPTICAL

Vision

Optical

ENVIRONMENTAL DESIGN

Thermal

Rain Removal and Defog System

ELECTRICAL/ELECTROMAGNETIC DESIGN

STRUCTURAL DESIGN

Pressure Loading

Canopy Latching System

Hail Impact

BIRD IMPACT DESIGN

MAINTENANCE PROVISIONS

Maintainability

Interchangeability and Replaceability

GENERAL ASSESSMENT

Ideally the most perfect crew compartment enclosure would be one that was spherical in shape so that the pilot would be afforded adequate protection from the environments; and yet, when he turns his head his visibility would always be normal to the surfaces. Materials have not existed that could be adequately formed to meet these requirements and at the same time have a reliable service life and acceptable aerodynamic drag.

Since polycarbonate has become available as a transparency material. General Dynamics has utilized this material in their design of a one-piece canopy in an attempt to obtain an ideal design. Aerodynamic and aircraft performance constraints undoubtedly were deciding factors in the outcome of the final shape.

Polycarbonate is highly susceptible to attack by chemicals and other environmental elements. As a consequence, the material must be either coated or other materials must be laminated to the exterior surfaces to provide necessary protection.

Previously, Douglas tested, under laboratory conditions, several coatings, and have found these coatings to be unsatisfactory. Generally, these coatings check or crack under temperature/pressurization conditions, and, in flight, were affected by other elements that ultimately turned the transparency transluscent. The F-16 windshield has a Texstar coating applied to the surfaces to provide protection. This coating has previously been applied to the Douglas DC-10 Wing Tip Lens Cover. To date, on a few covers that have been monitored, it appears that the coating must be either refurbished or replaced after about 4000 hours of service. This requirement was established because the coating becomes eroded or ablated due to aerodynamic effects. This cover is subjected to high and low temperature extremes that might approximate the F-16 conditions, but the unit is non-pressurized. Probably, the service life of the F-16 coated windshields can only be determined by monitoring canopies during the flight test and in-service program. Whether or not the coated monolithic transparency is more cost-effective than a laminated transparency has not been determined.

To date, Sierracin has formed both 3/8 inch and 1/2 inch thick monolithic canopies for application to the F-16 aircraft. Texstar has formed both 1/2 inch, 5/8 inch, and 3/4 inch monolithic canopies for application to the F-16 aircraft and bird test programs.

The transparency installation to the canopy frame is made with wet sealant. This approach is messy, time consuming to accomplish, and requires ideal temperatures above 70°F for 72 hours. A dumb-bell type seal was designed by Douglas for "dry-seal" installation that will greatly reduce the installation time for the transparency. The seal between the structural fairing and the transparency was replaced with a "dry-seal" (see Section XI). The dumb-bell seal concept has been successfully used by Douglas on the C-133, DC-8, DC-9, DC-10, and YC-15 aircraft windshields. Boeing, on some of their commercial aircraft, and Rockwell International, on their B-1 aircraft, have also used the dry-seal concept for windshield installations.

VISION/OPTICAL

Vision

General Dynamics (GD) has established vision diagrams for fighter aircraft based on the F-16 structural arrangement and its aerodynamic requirements. These vision diagrams, shown in Figures 6, 7 and 8, as supplied to Douglas, conform to the requirements of MIL-STD-850B for fighter aircraft except that the F-16 down vision at 20 degrees azimuth is 15 degrees instead of 20 degrees. The F-16A conforms to MIL-STD-850B as shown in Figure 7 and as described below:

1. At zero through 135 degrees azimuth, 90 degrees up vision is provided. At zero to 20 degrees azimuth, 15 degrees down vision is provided, increasing gradually to 28 degrees down vision at 30 degrees azimuth.

2. The vertical angles of clear vision in the area between 30 degrees and 90 degrees azimuth increases gradually from 28 degrees down to 40 degrees down at 90 degrees azimuth.

3. The vertical angle of clear vision will decrease from 40 degrees down at 90 degrees azimuth to 23 degrees down at 135 degrees azimuth.

The F-16B configuration conforms to MIL-STD-850B as shown in Figures 7 and 8 and as described below:

1. At zero degrees azimuth, down vision is 3 degrees up instead of 5 degrees down. Up vision of 12 degrees is provided.

2. At 20 degrees azimuth, 12 degrees down vision instead of 20 degrees is provided.

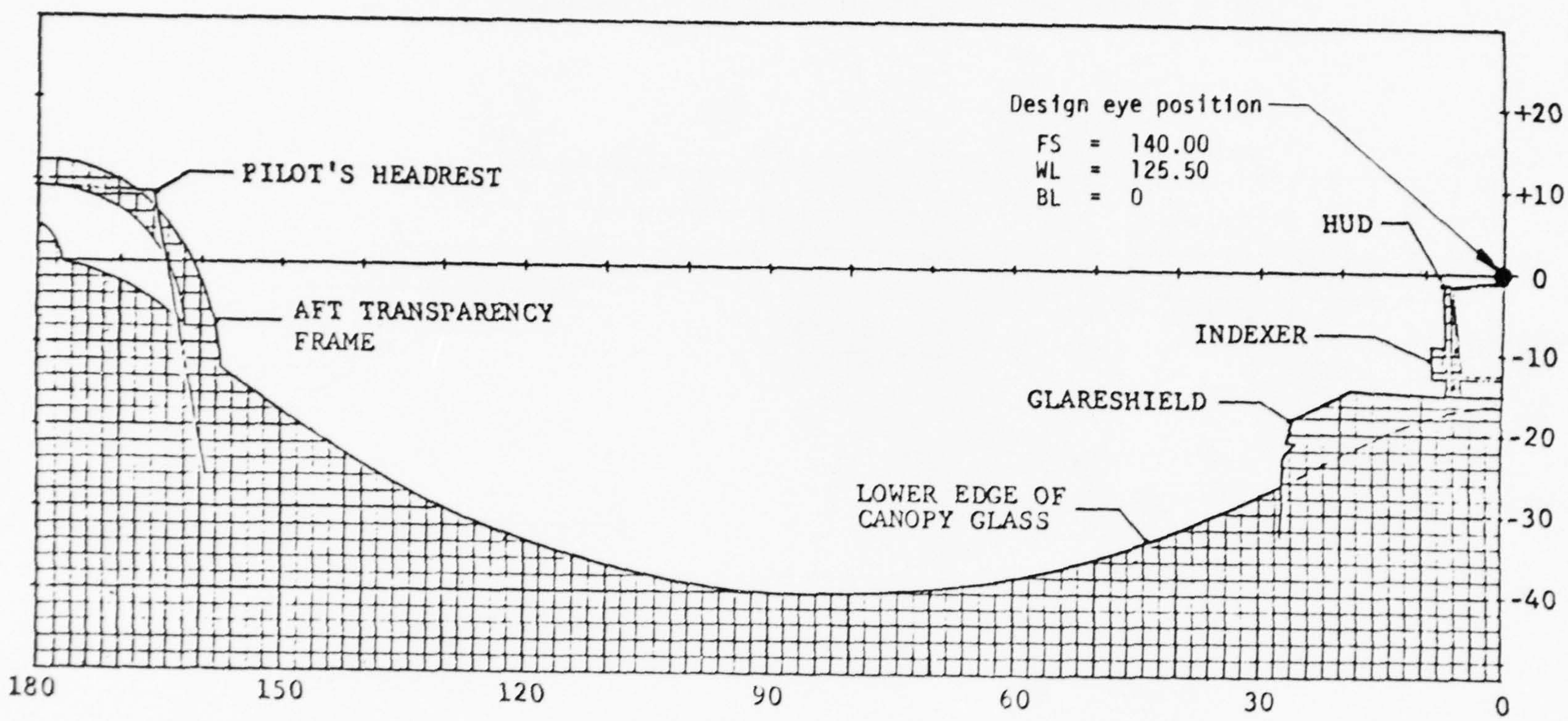

Figure 6. Pilot's External Vision (F-16A).

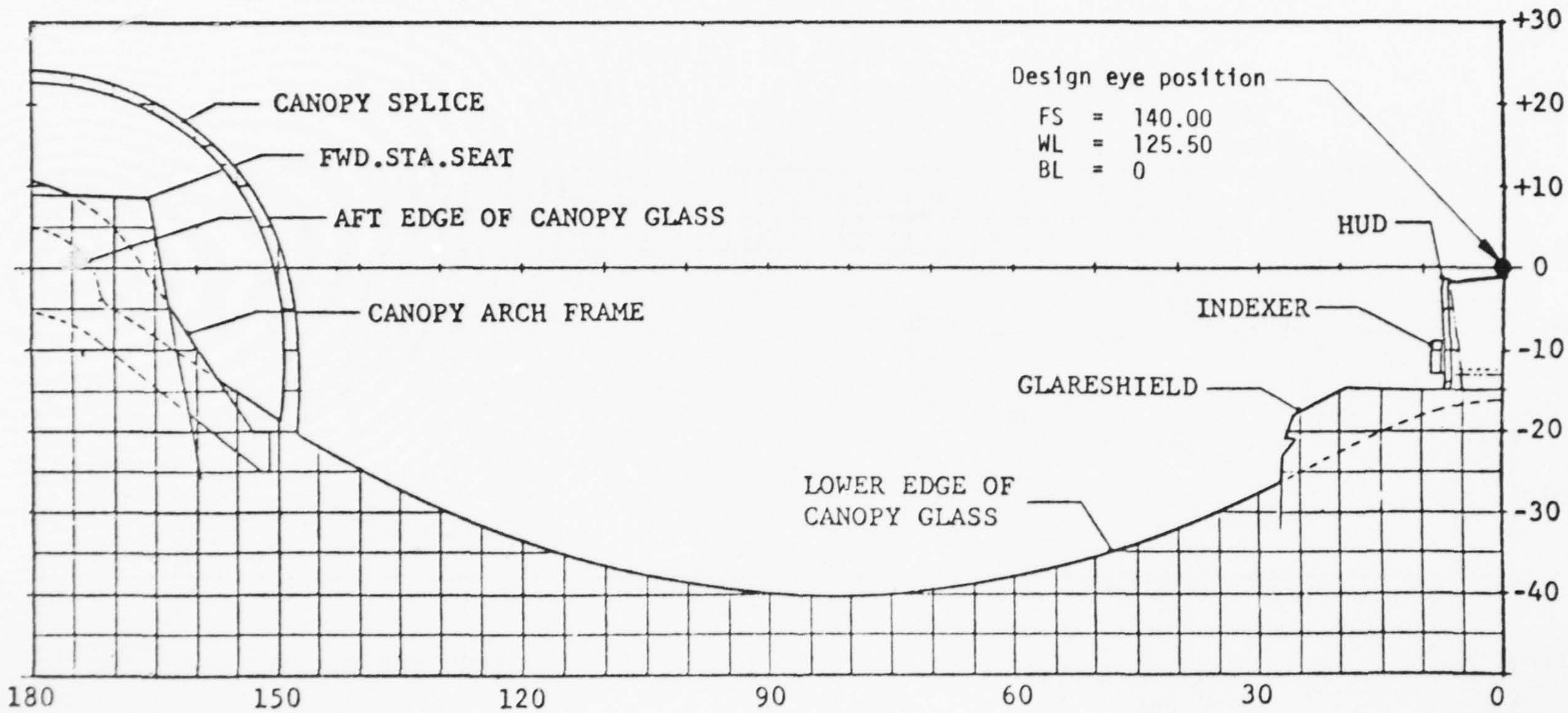

Figure 7. Pilot's External Vision - Forward Flight Station (F-16B).

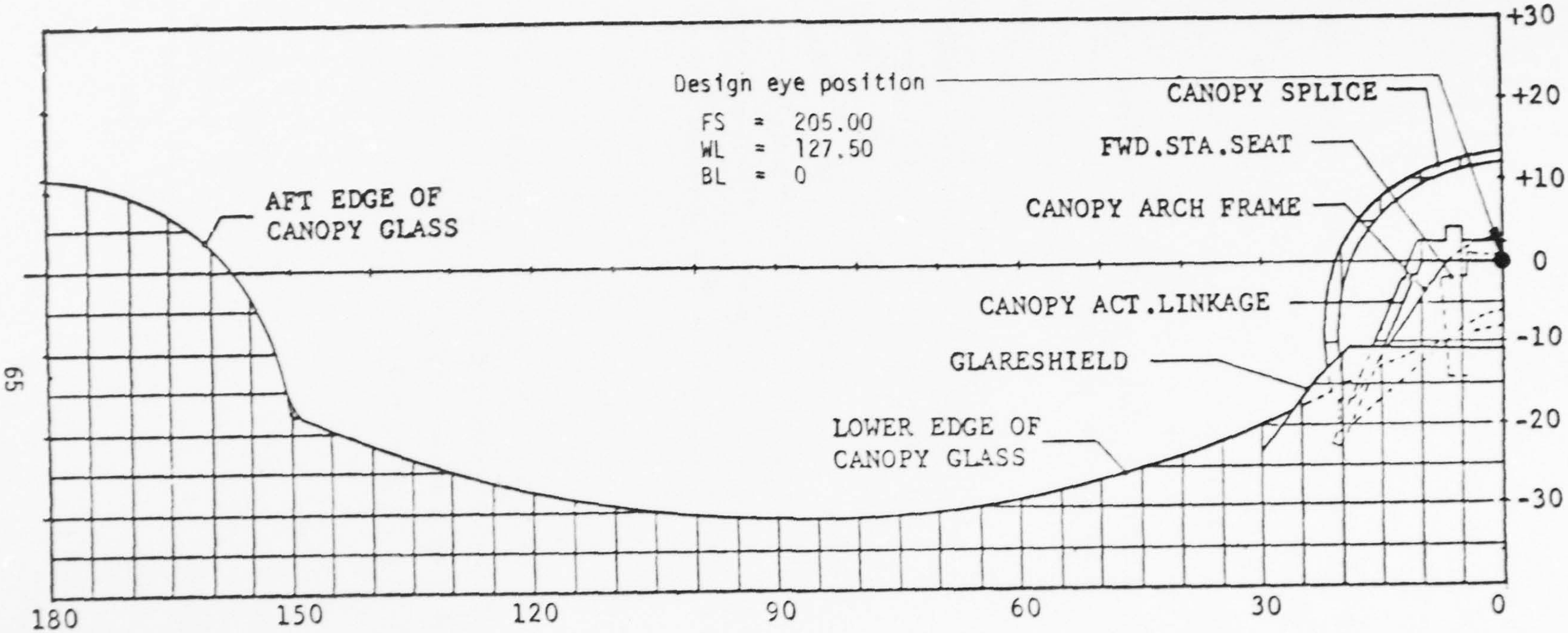

Figure 8. Pilot's External Vision - Aft Flight Station (F-16B).

3. At 30 degrees azimuth, 20 degrees down vision instead of 25 degrees, increasing to 40 degrees down vision at 90 degrees azimuth is provided.

4. The vertical angle of clear vision will decrease to 23 degrees down vision at 135 degrees azimuth.

The extent of the down vision in the aft flight station would be increased by moving the design eye position outboard as is allowed by MIL-STD-850B; however, the exact amount of improvement could not be calculated from the information available.

Head Clearance

The Air Force Design Handbooks delineate pilot head clearance to the inside of the canopy. The current F-16 design violates the 10-inch minimum radius head clearance envelope specified by AFSC DH 2-2 (DN2A1) by approximately 2 inches. It must be pointed out, however, that aerodynamic shaping requirements to meet the aircraft performance characteristics might have dictated this shape. In the event that a change in shape would not cause an aerodynamic problem, it is highly recommended for future investigation.

Alternate Design

As a portion of the Windshield Technology Demonstrator Program, investigative studies were conducted regarding a laminated canopy, the addition of a bow frame, and a fixed windshield forward of the pilot. These design concepts are addressed in Section XI.

Optical

The F-16A/B cockpit visibility for the pilot, of 15 degrees down over the nose and 40 degrees down over the side, is excellent. In the forward direction, the only obstruction above the 15 degree down line is the support structure for the gunsight combiner plate. Using a static

nominal left and right eye position, this structure will tend to obstruct one eye when viewing approximately 6 degrees to the left or right. In actual practice the pilot does not maintain a static position for his eyes, except possibly when receiving a reticle display from the gunsight, and will tend to look around this structure.

Elimination of the gunsight combining plate is feasible by projecting the reticles on to the canopy surfaces. This method entails a redesign of the display unit optics, the computational logic and possibly the use of special coatings on the canopy surface. This approach is further complicated because of the double curvature shape of the canopy. Therefore, this approach is not recommended because of the large development and design costs involved.

The F-16A/B canopy design consisting of a single, relatively thin ply provides a high luminous transmission capability. The double curvature shape, unfortunately, provides a built-in geometrical deviation for all light rays entering the cockpit. Visual sightings through the transparency are generally not affected if the deviation is less than approximately 10 minutes of arc (0.003 milliradians) for binocular vision of an object. As noted in Section II, a change in slope of the inner surface of 0.12 degree with respect to the outer surface produces the above deviation for the 3-degree up vision angle. Although this deviation was computed for flat surfaces, a comparable deviation will occur when the entrance and exit normals are nonparallel due to the double curvature.

Adding a gunsight combining plate produces a requirement for tighter deviation tolerances. The F-16A/B requirement for 0.003 milliradians maximum and 0.001 radian RMS, over an area defined as ± 6 degrees in azimuth, 3 degrees up and 15 degrees down may not be achievable over the entire area due to shape geometry and when measured for each eye rather than a nominal cyclops eye.

ENVIRONMENTAL DESIGN

The environmental design considerations include thermal design of the canopy, rain removal, and defog system.

Thermal

Reference 3 states that the maximum temperature for the forward portion of the F-16 canopy is a soak at 160°F followed by 10 minutes exposure to 290°F. This condition produces a temperature of almost 190°F on the inboard surface of the canopy at the end of the 10 minutes. MIL-E-38453A states that inboard surfaces of aircraft that are accessible to the occupant must be maintained below 160°F. This dictates that the canopy must be of a thicker material than the present canopy.

Rain Removal and Defog System

The rain removal and defog systems designed for the F-16 appear to be adequate to perform the job intended.

ELECTRICAL/ELECTROMAGNETIC DESIGN

The present General Dynamics design of the F-16 canopy does not include any provisions for electrical anti-icing, p-static control, lightning protection, RCS control and EMP control The latter two areas are not covered because SPO requirements for their inclusion do not exist. The General Dynamics decision to not include p-static control and lightning protection appears to be consistent with the design objectives of the aircraft, and this decision is substantiated by test data and experience on similar types of aircraft.

Although the investigations of this canopy design optimization program support the prior F-16 program decision not to apply lightning protection to the canopy, there are two areas of lingering concern. The first relates to possible electric shock from corona current immediately preceding and during a lightning strike. The second relates to the effectiveness of the grounding path to the fuselage when surface current discharges over the canopy to the fuselage.

The McDonnell Aircraft Co., tests (Reference 6) and opinions of other lightning effects specialists indicate that the corona current is of insufficient amplitude and duration to cause a real electric shock hazard. However, the reflex reaction to even a minor electric shock might have a detrimental effect on a pilot when he is already under high stress due to the problems associated with flying in a thundery environment. The determination of whether this concern is a practical problem is beyond the scope of canopy design. It is raised here so that the Air Force may conduct further investigation if it appears to be warranted. There are electrical design measures which are believed to be capable of reducing the internal electric fields to significantly reduce the possibility of electric shock. However, these design measures do not appear to be necessary, on the basis of present information.

The current path between the canopy surface and the fuselage appears to be random and perhaps unplanned. Since the current during a lightning strike can reach peak values of 200,000 amperes or more, the acoustic pressure wave and the electromagnetic field forces can be very high. The possibility of unacceptable damage from these forces appears to need special investigation, including actual testing of the canopy and interfacing aircraft structure. The electromagnetic interference (EMI) possibilities of the high discharge currents should also be investigated during EMI evaluation of the aircraft.

The present evaluation and test program for the F-16 (and most aircraft) does not have a category wherein the lightning originated electric shock hazard to the crew is evaluated. Past industry test approaches have concentrated on high voltage testing of canopies. Laboratory equipment limitations appear to have inhibited the evaluation of the potentially hazardous effects resulting from the very high magnetic fields which might be present in a cockpit when real lightning flashes across a canopy. It is strongly recommended that test limitations be overcome and that extensive analytical and test evaluation of the shock possibilities be investigated. If actual problems are confirmed, corrective action should be taken immediately.

STRUCTURAL DESIGN

A review of the F-16A Canopy Stress Analysis, General Dynamics' Report 16PR263, (Reference 9), would indicate that adequate design and positive margins-of-safety exist in the canopy support structure and hardware.

Pressure Loading

Load factors for design of the F-16 canopy are defined as follows (Reference 9):

1. Limit: The pressure differential between pressurized portions of the structure and the ambient atmosphere shall be 1.33 times the maximum attainable combined with the loads due to ground pressurization tests.

2. Ultimate: The limit-to-ultimate load factor is defined as 1.5.

These load factors are considered to be adequate for design of the F-16 canopy.

The General Dynamics' F-16 Transparencies Critical Item Development Specification, 16ZK002A (Reference 15), specifies the following loading conditions:

1. *Ultimate internal pressure:*

   13.2 psi at 265°F outer surface temperature
   and 190°F inner surface temperature. (Hold pressure
   for 5 seconds).

2. Ultimate external pressure

    8.5 psi at 245°F outer surface temperature and 150°F inner surface temperature. (Hold pressure for 5 seconds).

3. Internal cyclic pressure

    200 cycles at 8.0 psi, maximum temperature
    200 cycles at 5.7 psi, minimum temperature
    600 cycles at 8.0 psi, room temperature.

The General Dynamics F-16A Canopy Stress Analysis, 16PR263 (Reference 9), specifies the following loads as noted in Section II of this report.

5.10 psi maximum internal pressure plus tolerance on regulator valve.
6.80 psi limit maximum internal pressure
10.2 psi ultimate maximum internal pressure.

Reference 9 also states that the design loading on the transparency is internal pressure and that the maximum internal pressure for the canopy closed condition is 12.45 psi ultimate.

The discrepancy between the 13.2 psi noted in the Reference 16 and the 12.45 psi noted in Reference 9 is not explained.

As noted in Section II of this report, Douglas calculated 2000 hours as the service life for an F-16 canopy and 1814 cycles as one canopy lifetime. The temperature varied from between 290°F and -80°F. The loads stated above for the cyclic requirements appear to be conservative.

The margins of safety for the stresses due to pressure loading were determined by a series of tests (Reference 9). These tests were both static and fatigue and were conducted at 225°F, ambient, and -20°F temperatures. The test loads were 383 lb/inch design load and 575 lb/inch ultimate load. These loads exceeded the requirements established by

General Dynamics. The edge attachments were identical to those used on the actual airplane. No failures were experienced according to Reference 9.

Canopy Latching System

It was not clear from the materials received by Douglas whether any consideration was given to the deleterious effects of wear in the actuation and latching system. MIL-A-008866B (USAF) states the following:

> "3.2.8 Wear Endurance. Excessive wear of structural components . . . which would interfere with function of the part shall not occur . . ."

It appears that the design of the latching and actuation systems will meet the requirements of MIL-A-008866B. The detail parts in the systems that are subject to wear are made from PH 13-8 MO stainless steel material. Only in-service experience will determine the wear endurance of the various components of the systems.

Hail Impact

It was not clear from the data received by Douglas whether any consideration was made of possible canopy problems due to hail impact. Military Specification MIL-A-008866B (USAF) states the following:

> "3.2.9 Other Durability Considerations. The contractor shall develop and apply criteria for other durability considerations such as foreign object damage and special environments such as runway debris, sand, gravel, rain, hail, and lightning strikes. These considerations can arise due to airplane configurations, operations and substandard runways, or special atmospheric conditions. The criteria for these durability considerations shall require approval by the procuring agency.
>
> 6.2 Definitions.

> 6.2.1 Durability. The ability of the airframe to resist cracking, . . . corrosion, thermal degradation, delamination, wear, and the effects of foreign object damage for a specified period of time."

The Douglas recommended design criteria considering hail impact is presented in Section II of this report.

BIRD IMPACT DESIGN

Previously, General Dynamics performed bird impact tests on 0.50-inch monolithic polycarbonate transparencies (Reference 16). The testing was conducted at room temperature and the maximum speed tested was 350 knots with a four-pound bird. The impact point was eye level, centerline. The transparency deflected approximately 7 inches over the pilot's head.

The bird impact tests conducted as part of this program and described in Section VII of this report, proved that the current F-16 coated monolithic 0.50-inch polycarbonate transparency was not capable of defeating a four-pound bird at an impact velocity of 350 knots. The maximum obtainable impact velocity was established by tests to be slightly higher than 165 knots at a temperature of 75°F. The minimum thickness required to sustain the impact of a four-pound bird at Location A for 350 knots, and limit the deflection of the transparency above the pilot's head to less than 2 inches, is 0.76 inch, as determined from the test data presented in Section VII.

MAINTENANCE PROVISIONS

Generally, the concept of maintaining an aircraft canopy such as the F-16 canopy must be an all inclusive effort to define all potential maintenance items. The maintenance items that should be considered include:

- Removal and Replacement of Canopy
- Removal and Replacement of Transparency in Structure

- Servicing Mechanical and Electrical Components
- Cleaning and Repairability of the Transparency
- Preflight Checkout
- Postflight Checkout

Considerations for these items must be defined in terms of both maintainability and interchangeability.

Maintainability

It appears that General Dynamics has covered the most important maintenance items, but, in several different documents. These documents should be reviewed and combined into one documentation in the interest of saving time in either a shop or on the flight ramp.

Since the Air Force has become more aware of the costs of maintaining aircraft, cost of parts and turn-around time for installations, Douglas recommends dry-seals to replace the wet sealant applications between the canopy structure and the transparency, at two places, to reduce the maintenance man-hour installation time.

The monolithic polycarbonate canopy must have a protective coating on exposed surfaces. During pre-flight the transparency must be cleaned by experienced crewmen, exercising care, so that the protective coating is not damaged. Scratch damage tolerances must be established for the coating that will provide guidance in making a determination when the transparency needs repairing. It is recommended that General Dynamics and Texstar develop repair kits that would be available to crewmen. The repair kits should include instructions for applying material to cover the scratches; and thus provide the protection required for the polycarbonate. Subsequently, a determination would have to be made regarding pilot acceptance of numerous repaired scratches and erosion from a visibility/optical standpoint. A procedure should be developed regarding the degree of erosion and/or scratch repairs that are acceptable for continued flight versus removing

the transparency and recycling the unit back to Texstar for scratch removal, coating removal, and coating replacement. Probably the life of the part would have to also be considered in a cost-effect manner versus the remaining expected life of the polycarbonate after refurbishment.

Interchangeability and Replaceability

It is assumed that General Dynamics has demonstrated interchangeability requirements between the canopy structure and the transparency. The drawings for the respective parts have been reviewed, and it was found that each part has oversized holes for the attachments that should assure interchangeability.

An F-16A transparency replacement time study was performed during the bird impact tests reported in Section VII.

Two methods of replacement were studied. The first method consisted of removing the transparency/canopy unit from the test fixtures (which simulated the aircraft) and then removing the transparency from the canopy frame. It required four men (General Dynamics personnel) 6 hours elapsed time to replace the transparency and install the canopy/transparency on the test fixture. This time did not include curing time for the wet sealant required to install the current F-16A transparency. This system of replacement involves the mechanisms, latches, actuating system, alignment problems, and clearances on the canopy latches.

The second method of transparency replacement was to remove the transparency from the canopy frame without removing the canopy frame from the test fixture. This method required two men and two hours elapsed time. Leaving the canopy frame on the aircraft eliminates disconnecting the actuating system and reduces alignment problems.

The time goal for a complete transparency removal and replacement on an operational aircraft is three hours elapsed time for a two-man team using standard tools.

SECTION IV
TEST PLANS AND REPORTS

The Windshield Technology Demonstrator Program is a generic program aimed at reviewing all aspects of transparency design concepts, systems requirements, maintenance, reliability and cost of ownership with direct application to at least two Air Force aircraft.

The program was an all-encompassing effort directed toward specific objectives:

- To develop criteria and requirements for the design and evaluation of total aircraft windshield and canopy systems.

- To perform well-defined and comprehensive studies, tests and analyses of windshield and canopy designs, structures, systems and materials.

The end result of the program was the documentation of meaningful data that is beneficial to the advancement of the various technologies associated with the design of aircraft crew compartment transparencies and supporting structural members. Contained in this report are the results of studies and analytical evaluations of test data directed toward canopy design with direct application to the F-16 canopy.

Supplementing this report are a series of reports that have been formulated. A succinct description of each is presented in subsequent paragraphs. Each report has been issued by the Air Force as a Technical Report with "Unlimited Distribution" to industry.

AFFDL-TR-76-75 (MDC J6952) EFFECTS OF LABORATORY SIMULATED PRECIPITATION STATIC AND SWEPT STROKE LIGHTNING ON AIRCRAFT WINDSHIELD SUBSYSTEMS

This four section report contains the test plans, test results, conclusions, a summary and recommendations of a program that evaluated

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candidate outer ply material for aircraft windshields and canopies. These materials were subjected to laboratory simulated static electric charging and swept stroke lightning tests.

The potential for possible aircraft problems increases with increasing physical outer surface area of the windshield or canopy. Very high values of static charge can accumulate on the outer insulating surfaces. When the charge can no longer be contained, electrical discharging occurs, usually in the form of surface flashing. The discharge generates electrical transients which causes electromagnetic interference and possible damage to electrical components, depending on the system and component design.

Methods of generating the electrostatic charge for test purposes and the control and evaluation of the discharges are covered.

Potential damage to aircraft windshields or canopies from swept stroke lightning was also evaluated by subjecting test samples to simulated lightning.

Neither the static electric tests nor the lightning tests resulted in fracture or catastrophic destruction of the outer ply test samples. Dielectric puncture of the samples was not evident, even though the test environment was very severe. Minor surface etching due to static discharges was noticed on some of the specimens while more severe surface etching of all the glass specimens was caused by the lightning tests.

Recommendations are included which should enable the design of windshield and canopy systems that are more immune to the effects of precipitation static charging and swept stroke lightning.

A study of the effects of electromagnetic pulse (EMP) in windshield design was added as a subsequent study effort and the results are included in an appendix to this report (AFFDL-TR-76-75).

AFFDL-TR-76-114 (MDC J7173) THE DETERMINATION OF DEFLECTION AND STRESS DISTRIBUTION FOR A LAMINATED TRANSPARENT BEAM

This report documents the theory and procedures for calculating the elastic deflection and stress distribution of a flat, laminated beam comprised of transparent materials, with fixed ends or simply-supported ends, subjected to a normal point load at the midpoint of the beam. The development of a computer program using the basic theory is presented along with illustrative examples and a "user's manual".

One of the basic objectives of the Windshield Technology Demonstrator Program was to develop windshield design technology for application to high performance military aircraft. Pursuant to this effort, a need arose for a simplified analytical method of rapidly assessing the structural effectiveness of candidate laminate configurations which consist of alternating structural plies and interlayers. Structural plies are composed of materials such as polycarbonate, glass or acrylic; interlayers are composed of materials that may be less stiff by several orders of magnitude.

A realistic method of static analysis for a laminated beam carrying a concentrated load at the center, representing a bird impact, and having various types of end supports, would be a useful tool. Such a method should avoid the frequently made Bernoulli-Euler assumption that plane sections remain plane during deformation, because shear deformations in the soft interlayers are large and must be considered. The significance of these deformations is shown experimentally in Reference 1

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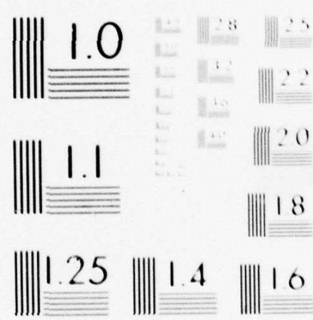
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of the report. Other rough assumptions that would unnecessarily diminish the effectiveness of the analysis should also be avoided. The method should accommodate laminates composed of nine layers or more, with different material properties for each layer.

Apparently no method meeting these requirements exists in the literature, although related work is described. Reference 2 noted in the report contains an analysis of a three layered beam based on the assumption that the angle of rotation between a cross section of the center layer and its undeformed position is a constant factor times the slope along the length of the beam. This assumption is considered unnecessary, and its effects on the reliability of the results are hard to assess. The solution is based on the energy method, which introduces further approximations. Reference 3 noted in the report presents methods of analysis of multiple layered beams based on the Bernoulli-Euler hypothesis, consequently, the approach is not applicable to beams with soft interlayers. Reference 4 noted in the report offers an analysis of three layered beams involving simplifying assumptions which were also not considered acceptable for the present application.

Therefore, a new approach to the problem was developed. The approach is based on the assumption that each structural ply can be treated as a beam to which the Bernoulli-Euler hypothesis is applicable, but no such assumption is applied to the cross section as a whole. Structural plies can bend and stretch, but shear deformations of these plies are considered negligible. Interlayers are assumed to carry in-plane, or transverse shear, but not axial loads or bending moments. Deformations through the thickness of the beam are considered negligible, but stresses normal to the layers are assumed to exist. The equations of equilibrium and compatibility are written. The resulting set of differential equations is then expressed as a single matrix differential equation which is solved exactly.

The analytical results have been translated into an efficient Fortran program. This program applies to nine layer laminates, which is an adequate number in most cases. The program can be easily extended to cover more layers if necessary. Fewer layers can be accommodated by introducing negligible stiffness properties (E and G) for some of the layers. The method applies to fixed ended beams, but the program can be modified to be applicable to other boundary conditions. It can be applied to a beam with pinned ends by considering a fixed end beam twice as long as the beam under consideration. The data output by the computer program between the quarter points of the fixed end beam is applicable to the pin end case. Extension of the method to other loading conditions is possible.

The most significant simplification involved in the analysis is the assumption that deformations through the thickness are negligible. This assumption greatly simplifies the analysis without slighting the primary feature of windshield laminate behavior, which is the relative freedom of structural plies to slide past each other because of the low stiffness of interlayer materials. The only negative effect of the assumption is that stresses normal to the layers are not correctly predicted. This is believed to be a localized effect confined to the center and ends of the beam where loads are applied. The elimination of this assumption is a possible subject for additional research. An analysis which accounts for transverse deformations might provide data that would be useful in defining adhesive strength needed to prevent delamination in regions of high transverse loads.

Results of the analysis have been correlated with finite element results and test data as described in the report (AFFDL-TR-76-114). The comparisons are good and verify the validity of the basic assumption.

AFFDL-TR-76-156 (MDC J6944) DAMPING, STATIC/DYNAMIC, AND IMPACT CHARACTERISTICS OF LAMINATED BEAMS TYPICAL OF WINDSHIELD CONSTRUCTION

This eight section report contains the test plans, test results, and analyses for the damping, static/dynamic, and impact beams, computer math model analyses and results, and conclusions formulated for various monolithic and laminated transparent beam specimens.

The objectives of this test program were as follows:

1. To develop effective instrumentation and testing techniques.
2. To evaluate the application of strain gages for the determination of energy transfer between the specimen laminates.
3. To test representative windshield laminated structures in the form of uniform straight (flat) beam sections.
4. To determine the damping characteristics, static/dynamic responses, and impact responses of the specimen beams under different temperature, edge restraints, and loading conditions.
5. To use the beam parameters calculated from the test data in a cantilevered-beam computer math modeling program.
6. To compare the results of the test program with the results of the computer analytical program in order to verify math models.

The method of data collection was assessed for data reduction that could provide stress-strain relationships. These relationships would be used for the determination of energy transfer, dynamic response, and displacements due to specified loads.

The conclusions derived from the three major types of testing were:

Damping Beam Testing

1. These tests provided an effective means of gathering the required damping parameters. These tests also provided a data base for comparison with analytical results.

2. The Damping-To-Stiffness Ratio, h, provides an effective means for comparison of transparent windshield materials.

Static/Dynamic Beam Testing

1. The simply-supported laminate beams, with bolts and bushings, present a higher equivalent beam stiffness, EI, than those with the fixed-end conditions. The attachments prevent structural ply slippage at the ends, thus providing a type of inner-fixity.
2. For the fixed-end beam configurations, the single-row attached beam has approximately an 84-percent fixity and the double-row attached beam has approximately 96-percent beam end fixity in respect to a theoretical fix-ended beam.
3. The specimens tested all have an approximately linear relationship with respect to applied loads, structural ply strains, and centerline deflections.
4. It appears that the strains within a particular structural ply vary linearly. Specifically, after deformation of the beam occurs, plane sections of individual plies remain plane.
5. Beam specimens with the CIP interlayers allow the structural plies to behave more independently; while the PPG-112 interlayer stiffens the beam, thus causing the specimen to behave relatively more like a monolithic beam.

Impact Beam Testing

1. This type of test was effective in providing the highest practical strain-rate in the beam specimens under laboratory test conditions. The strain-rate achieved was approximately 10 in./in./sec. This is within an order of magnitude of the range of maximum strain-rates expected (50 to 200 in./in./sec.) due to an actual bird strike.

2. The failures of the strain-gage system appeared to closely follow the structural ply failures.
3. Post-test observations showed the induced failures to be of a brittle-fracture type, typical of fractures found on canopies after bird impact testing.

AFFDL-TR-77-92 (MDC J7171) EVALUATION OF WINDSHIELD MATERIALS SUBJECTED TO SIMULATED SUPERSONIC FLIGHT ENVIRONMENTS

This report presents the test plan and the results of a series of wind tunnel tests which were designed to evaluate transparent plastic and glass materials for candidate aircraft windshields that must withstand the effects of aerodynamic heating and differential pressure which result from supersonic velocities at operating altitudes.

The goals of this test series were to simulate the operational environment of current Air Force inventory of supersonic aircraft and to determine the effect of this environment on selected face ply materials, to ascertain its effect on the optics of selected transparent materials, and to evaluate its effect on specific edge designs and coatings for a glass face ply.

The test specimens were flat and approximately 7 x 11 inches. They consisted of two general groups: monolithic acrylic and coated polycarbonate materials; laminated multi-ply polycarbonate composites utilizing various face plies and edge designs. A total of twenty-two (22) specimens were tested.

The test specimens were instrumented with thermocouples to provide a temperature-time history. A Hasselblad camera was mounted outside the tunnel to take sequence photographs through the specimens during the wind tunnel testing. Another Hasselblad camera was used to take pretest and post-test photographs so that the permanent optical distortion could be evaluated.

A realistic flight environment was chosen and used to calculate the test environment. Because of tunnel limitations it became necessary to revise these test conditions to be compatible with the tunnel capability.

The windshield materials were subjected to the modified test environment and an evaluation was achieved by appraising the face ply materials, by evaluating the optical distortion of the candidate windshields, both temporary distortion and permanent distortion, and by measuring the change in light transmission and haze in the transparent materials after being subjected to the modified simulated flight environment. The assessment of the edge designs for the glass face ply was incomplete and it was not possible to evaluate the coating on the glass ply.

This test series was completed in three days in the Hypersonic Wind Tunnel (b) of the Von Karman Gas Dynamics Facility (VKD) located at the Arnold Engineering Development Center (AEDC), Tennessee.

BACKGROUND

It is apparent to the designer of supersonic aircraft windshields that aerodynamic heating strongly influences a windshield design, since the transparent materials must be able to withstand a severe heating environment and still provide acceptable optics for the pilot. Specifically, it is immensely important to appraise the reliability of the face-ply material at the operational limits of the materials, and to determine what effect the aircraft operational environment has on the windshield optics. To properly evaluate these areas, a test plan was devised to describe a test environment comparable to actual selected flight conditions.

Comparative performance, reliability and capability information was desired for currently used stretched acrylics; for a windshield cross section comprised of an acrylic outer ply, an interlayer and a polycarbonate inner ply; for exterior protective coating on polycarbonate; for

precipitation-static coatings on glass; and for edge member concepts for glass face ply designs.

A previous experimental investigation, conducted by the Arnold Engineering Development Center provided a valuable source of information for this test series, but did not include a "cabin" environment and resulted in test panel deflections that one would not expect to be realistic. To provide realistic temperature and pressure gradients, a simulated cockpit enclosure was incorporated for this test operation.

AFFDL-TR-77-96 (MDC J6950) TESTING FOR MECHANICAL PROPERTIES OF MONOLITHIC AND LAMINATED POLYCARBONATE MATERIALS

This report covers the materials, test plans, test results, statistical methodology, data reduction, computer programs, correlations, and analyses, for determining the mechanical properties of monolithic and laminated polycarbonate materials.

Current industry tests and publications offer incomplete and/or inadequate information concerning high-performance material properties of many promising aircraft transparency materials. Such properties are essential to effect a proper balance of structural efficiency, safety, and minimum weight designs, and to provide a solid basis for specification control tests and design trade-off studies. Further, it was necessary that a sufficient number of tests be conducted in arriving at the desired material properties such that the values obtained are minimum guaranteed allowable properties for a material as processed by a specific fabricator.

Prior industry work in the determination of material properties was concerned with materials testing at low strain rates and moderate temperatures. Properties so obtained are utilizable when the critical design

conditions are based on pressure and thermal loading only, but are not accurate for a bird impact loading condition.

To be of maximum usage, the required material properties should be generated in a test program that considers effects on the material due to:

1. Rate of strain at appropriate testing temperatures
2. Effects of forming
3. Effects of prior thermal history of the material
4. Effects of fusion bonding
5. Effects of in-service heating and aging.

It was determined that eight series of tests were necessary to establish specific material properties of selected aircraft transparency materials. These tests determined the effects on the material properties when subjected to high and low strain rate loading and when exposed to a critical temperature-time condition. Material strength properties were determined from manufactured specimens, specimens taken from actual instrumented beam test samples and previously impacted windshield test parts. A series of candidate interlayer materials laminated and/or cast in various transparent glazing constructions were subjected to various optical tests. A representative section of the candidate windshield edge design was subjected to a critical cyclic loading condition at its maximum equivalent temperature, and also exposed to extreme thermal shock conditions to determine strength and material degradation.

AFFDL-TR-77-97 (MDC J7172) STANDARDIZED WINDSHIELD FABRICATION SPECIFICATION

This report covers the preparation of a Critical Item Product Fabrication Specification (CIPFS) for a windshield design. The report format was developed in accordance with MIL-STD-490 Appendix X.

The CIPFS document provides the performance, design, fabrication and testing requirements for the preproduction and production phases of producing a windshield design.

The performance requirements cover:

- Structural
  - Bird Impact
  - Pressure/Temperature Cycling
- Anti-static
- Anti-icing
- Optical
  - Transmission/Haze
  - Distortion/Deviation
  - Defects
- Electromagnetic Pulse
- Radar Cross Section Control
- Environmental
  - Temperature/Humidity/Fungus
  - Sunshine
  - Solvents/Solutions/Abrasions

The design and fabrication requirements c

- Reliability
- Maintainability
- Interchangeability
- Environmental
- Standards
- Materials
- System Interface
- Inspection and Acceptance

The testing requirements are basically in two categories; first, the preproduction phase that verifies that the design and fabrication of the windshield panel meets the requirements of the CIPFS; second, the production phase that verifies that the production articles will continuing to meet the requirements of the CIPFS.

AFFDL-TR-77-141 PRECIPITATION STATIC ELECTRICITY AND SWEPT STROKE LIGHTNING EFFECTS ON AIRCRAFT TRANSPARENCY COATINGS

Swept stroke lightning and static electric discharges on aircraft windshields and canopies will induce large electrical transient currents into the fragile transparent conductive coatings which are used for anti-icing, de-fogging, and radar cross section control of these transparencies. A test program was conducted to study possible detrimental effects of these transient currents.

Representative aircraft transparencies were subjected to laboratory simulated swept stroke lightning magnetic fields and static electric charging. Damage assessment methods included visual inspection with normal and polarized light, electrical resistance measurements of the conductive coating, and thermal inspection using a closed circuit infrared television technique.

The results indicated that swept stroke lightning with a high level re-strike will most probably cause a failure or instigate a failure in a heated coating during subsequent heating cycles. The tests also indicated that static electric induced transients in a moderately large transparency should not cause problems in coatings that are properly designed and applied. However, these same transients can initiate an ultimate failure if visible or invisible falws are present in the original conductive coating.

SECTION V
THERMAL DESIGN STUDIES

This section presents the results of thermal studies that were conducted to support a laminated canopy design for potential use on the F-16 aircraft. The basic cross section selected for this program was a three ply laminate consisting of an acrylic face ply and a polycarbonate structural ply joined by an interlayer. Studies were conducted to assist in the definition of material thicknesses, to determine the adequacy of the existing defog system, and to provide temperatures for an edge attachment area test program.

MATERIAL STUDIES

Material temperature studies were performed to: define temperature gradients at the most severe operating conditions, establish temperatures expected during a potential birdstrike, determine the highest rate of material temperature change, establish the minimum allowable canopy thickness for crew comfort as required by MIL-STD-38453A, assist in the selection of the interlayer thickness, and provide information for edge to frame clearance requirements.

Temperature Gradients

Temperature gradients were determined for two canopy conditions:

- Maximum and minimum temperatures.
- Maximum and minimum temperatures when the aircraft is operating below 8000 feet (AGL).

These studies were necessary to establish the material operating limits and the material temperatures for a potential birdstrike.

The thermal analyses for temperature distribution and birdstrike studies presented in this section are for a laminated configuration that consisted of a 0.875-inch core ply of polycarbonate, an external ply of

0.125-inch acrylic and 0.080-inch S-100 silicone interlayer, as shown in Figure 9. The acrylic face ply of 0.125-inch was used because it was the minimum thickness available in large sheets. The proposed concept in Section XI specified an 0.080-inch acrylic external ply which is the minimum acceptable thickness. The 0.875-inch thickness of polycarbonate was selected because it is the median thickness of polycarbonate utilized in the design concepts presented in Section XI. The polycarbonate thickness for the concepts presented in Section XI varies from 0.740 inch to 0.990 inch.

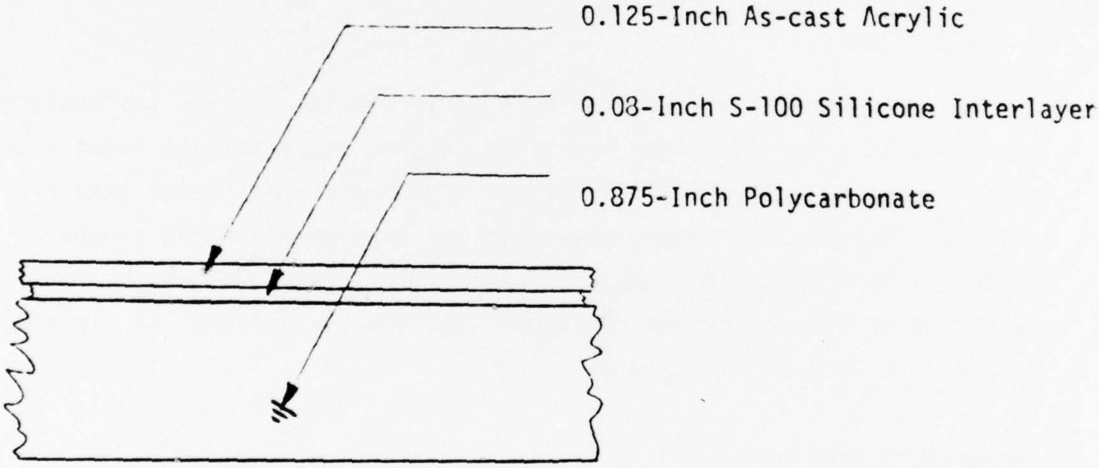


Figure 9. Cross Section of Polycarbonate Composite Canopy.

Maximum and Minimum Temperatures

This study was conducted to determine the temperature distribution within the candidate canopy when maximum or minimum temperature conditions occur. The maximum canopy temperatures occur subsequent to a 10-minute supersonic cruise in "U.S. Standard" atmosphere. The "U.S. Standard" atmosphere is defined in Table 2A-1 of Reference 17. The maximum temperature distribution is shown in Figure 10. The minimum canopy temperatures

occur during subsonic cruise with a -80°F ram-air temperature. The minimum temperature distribution is also shown in Figure 10.

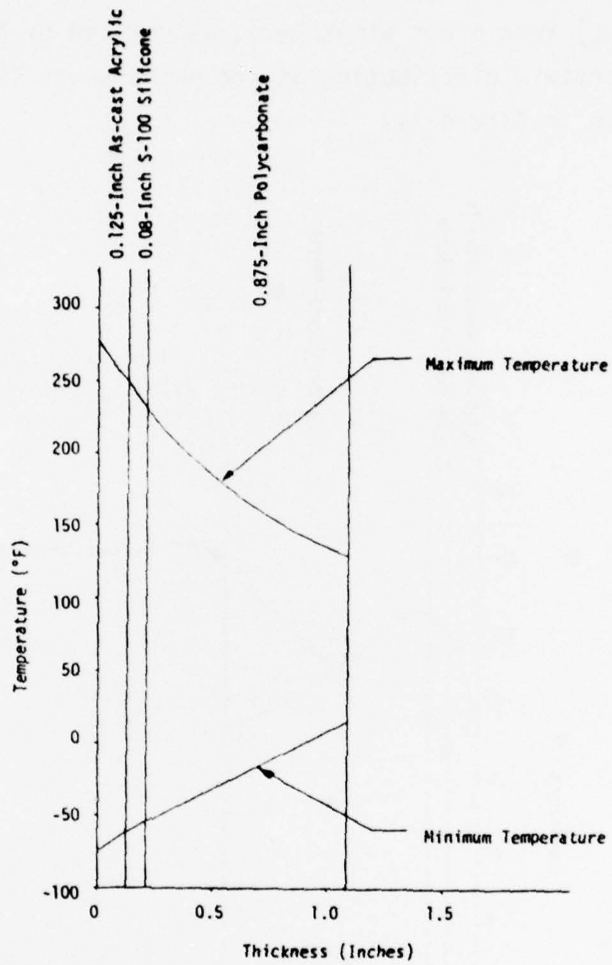


Figure 10. Maximum and Minimum Temperature Distribution Through Canopy.

Gradients for Maximum and Minimum Temperatures Below 8000 Feet

After the theoretical and operational thermal environment was developed for the F-16, studies were conducted to determine the probable maximum and minimum temperatures that the canopy could experience when the aircraft is at 8000 feet (AGL) or below. This is based on the premise that the probability of a bird strike above 8000 feet (AGL) is negligible.

The maximum bird strike temperature that the canopy could experience would be subsequent to an emergency descent to 8000 feet (AGL) or below from a 10-minute supersonic cruise and reduce speed to Mach 0.52. The cruise would be in a U.S. Standard atmosphere, and the descent would be to 8000 feet (AGL) into a hot atmosphere, as defined by MIL-STD-210B. The maximum temperature distribution at the point where the bird strike might occur is shown in Figure 11.

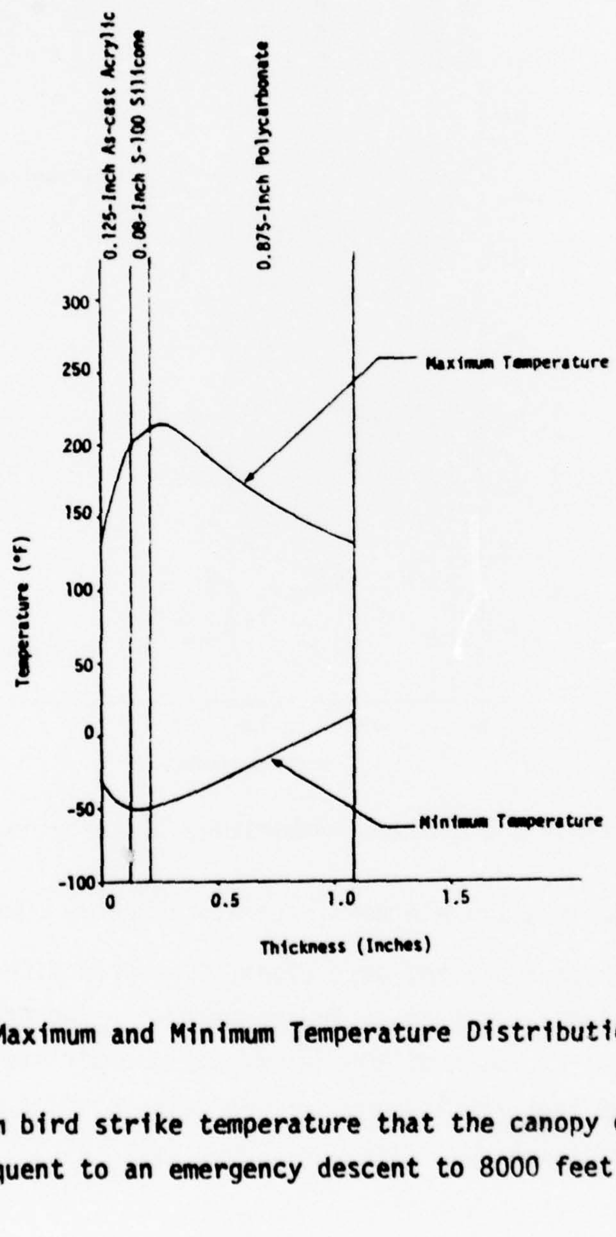


Figure 11. Maximum and Minimum Temperature Distribution for Birdstrike.

The minimum bird strike temperature that the canopy could experience would be subsequent to an emergency descent to 8000 feet (AGL) from a

subsonic cruise at -80°F ram-air temperature at altitude, as defined by MIL-STD-210B, and a speed of Mach 0.85. At 8000 feet (AGL) the temperature distribution at the point where the bird strike might occur is also shown in Figure 11.

Material Rate of Temperature Change

This study was conducted to determine the highest rate of temperature change that could be experienced by the canopy materials. The maximum-rate condition for a temperature change from hot to cold would be a supersonic cruise in a standard atmosphere, followed by a deceleration to subsonic cruise, with a ram-air temperature of -80°F. The resulting temperature profile is shown in Figure 12 for the configuration shown in Figure 9.

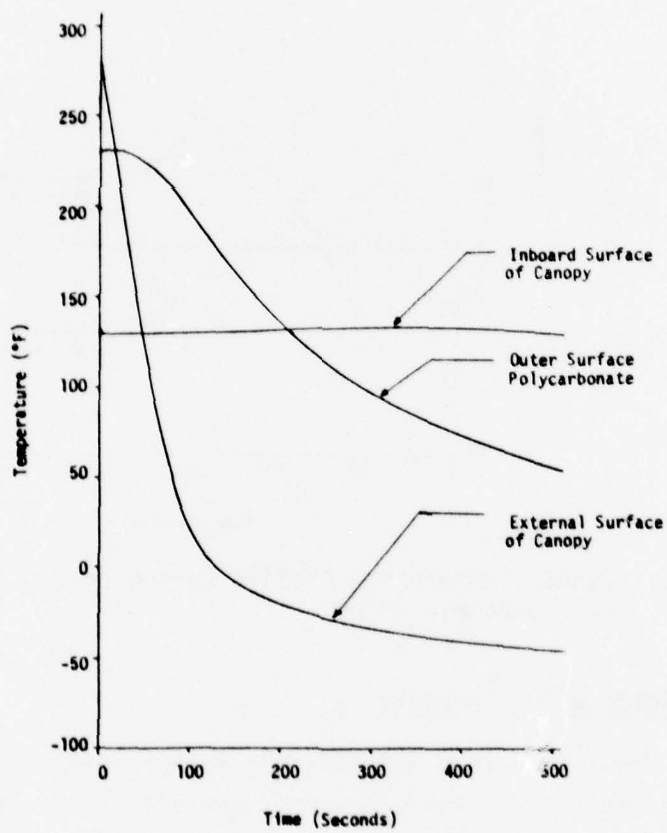


Figure 12. Canopy Temperature Profile During Deceleration from Supersonic Speed to Subsonic Speed.

The maximum rate condition for a cold-to-hot temperature change is a "soak" at subsonic cruise, with a -80°F ram-air temperature followed by an acceleration to supersonic cruise in a standard atmosphere. The resulting temperature profile is shown in Figure 13.

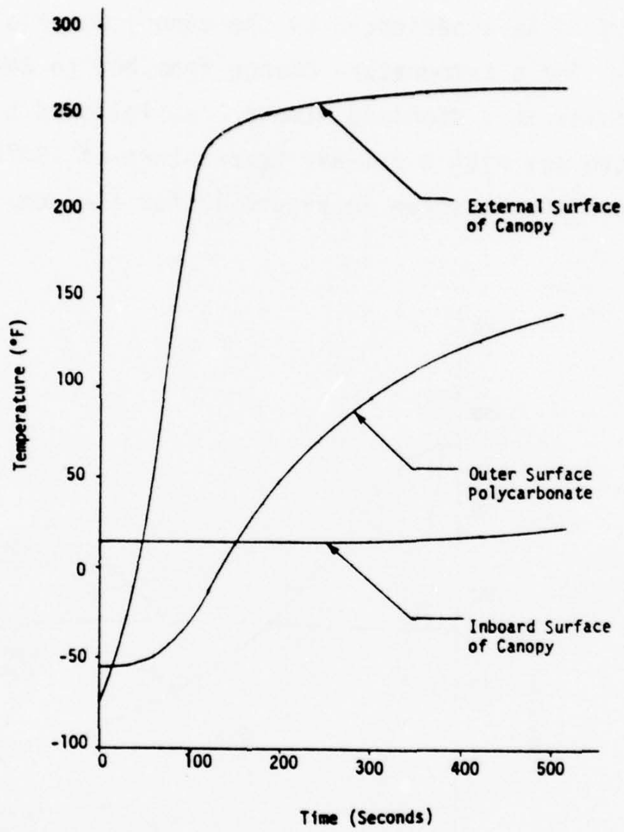


Figure 13. Canopy Temperature Profile During Acceleration from Subsonic to Supersonic Speed.

Minimum Thickness Requirements

It was determined from the analysis that a total thickness of 0.725 inch minimum was required to keep the inner surface of the transparency below the 160°F requirement established by MIL-STD-38453A.for crew comfort. The minimum acceptable laminate would then consist of a 0.565 inch polycarbonate core ply, an 0.080 inch acrylic face ply and an 0.080-inch silicone interlayer.

Table 4 presents a comparison of temperatures for four laminated configurations. These results indicate that the maximum inner surface temperature for the hot condition is only 7°F hotter for the 0.750 inch polycarbonate than for the 0.875 inch polycarbonate. These analyses were performed utilizing the 0.080 inch minimum thickness acrylic face ply.

TABLE 4. MAXIMUM AND MINIMUM CANOPY TEMPERATURES

Canopy Material	Max. Surface Temp. (°F)		Min. Surface Temp. (°F)		Average Birdstrike Temps. (°F)	
	External	Internal	External	Internal	Maximum	Minimum
1/2-In. Polycarbonate 0.08-In. Acrylic 0.080-In. Silicone	285	167	-75	0	190	-30
5/8-In. Polycarbonate 0.08-In. Acrylic 0.080-In. Silicone	285	154	-75	5	185	-30
3/4-In. Polycarbonate 0.08-In. Acrylic 0.080-In. Silicone	285	145	-75	10	180	-25
7/8-In. Polycarbonate 0.08-In. Acrylic 0.080-In. Silicone	285	138	-75	15	180	-25

Material Thermal Expansion

An investigation was made to determine the thermal elongation of a laminated F-16 canopy. The purpose of this analysis was twofold:

- To determine the shear strain in the interlayer.
- To determine canopy frame clearance

The differential was based on the 90-inch length of the canopy from front to back and half of the total expansion was applied at each end. Two assumptions were made to complete the analysis. They were: (1) The canopy is in equilibrium at 75°F. (2) The coefficient of thermal expansion for as-cast acrylic follows the curve in Figure 14 extrapolated for high temperatures.

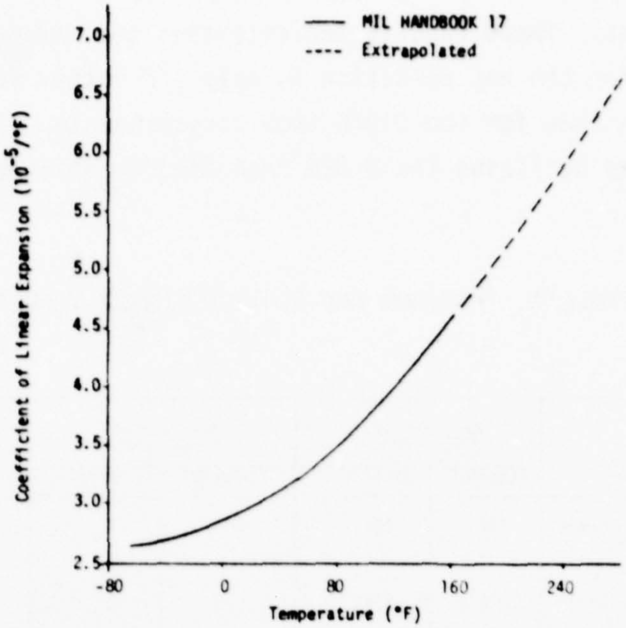


Figure 14. Effect of Temperature on the Coefficient of Linear Thermal Expansion of As-Cast Acrylic.

Interlayer Shear Strain

The selection of interlayer thickness is dependent on the shear strain in the interlayer caused by differential thermal expansion of the acrylic face ply and the polycarbonate structural ply.

Figures 15 and 16 show the maximum temperature conditions for laminates of 0.125-inch as-cast acrylic and 0.625-inch polycarbonate with an interlayer of silicone (Sierracin S-100) of 0.080-inch and 0.100-inch, respectively. The dashed line graphically shows the limits of thermal extension of an uninhibited interlayer and was based on half of the canopy length. The shear strain of the interlayer at the edge, from Figures 15 and 16, is 1.6 in./in. for 0.080-inch interlayer and 1.36 in./in. for 0.100-inch interlayer. The average allowable cohesion rupture strain for 0.120 inch silicone interlayer at 195°F, low strain rate, is 1.6 in./in. (Reference 18).

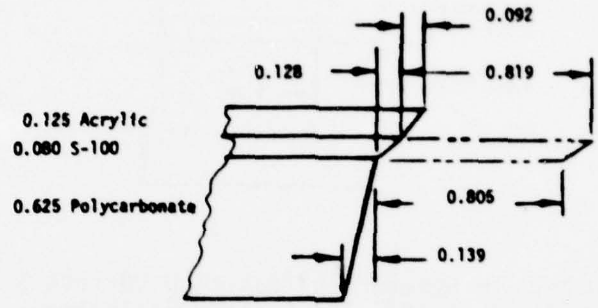


Figure 15. Maximum Temperature Effect on 0.080 Silicone Interlayer (Sierracin S-100) (Dimensions in Inches).

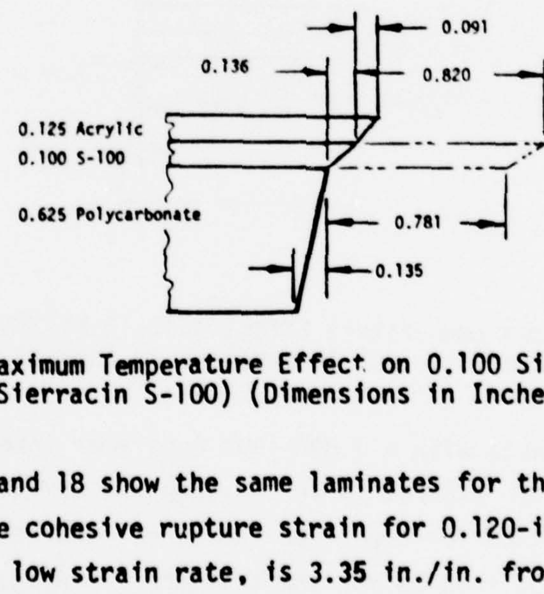


Figure 16. Maximum Temperature Effect on 0.100 Silicone Interlayer (Sierracin S-100) (Dimensions in Inches).

Figures 17 and 18 show the same laminates for the maximum cold conditions. The cohesive rupture strain for 0.120-inch silicone interlayer at -30°F, low strain rate, is 3.35 in./in. from Reference 18. The interlayer shear strain shown in Figures 17 and 18 is well below this limit.

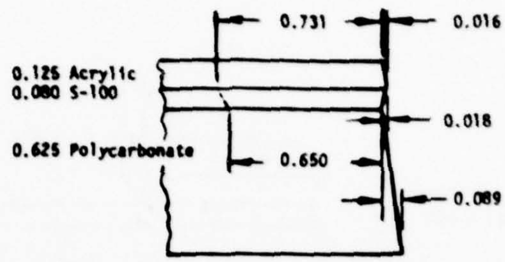


Figure 17. Minimum Temperature Effect on 0.08-Inch Silicone Interlayer (Sierracin S-100) (Dimensions in Inches).

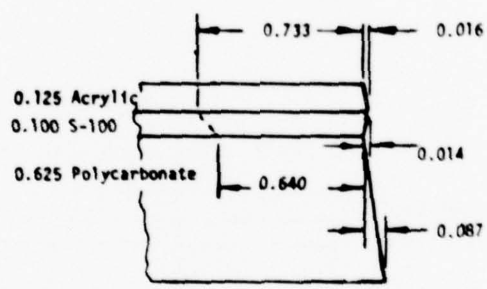


Figure 18. Minimum Temperature Effect on 0.10 Silicone Interlayer (Sierracin S-100) (Dimensions in Inches).

A similar laminate with a 0.030-inch copolymer interlayer (Sierracin S-130) was analyzed. Figures 19 and 20 show the maximum hot and cold temperature conditions for the S-130 laminate. The relative excursions are not as great for this as for the silicone laminate, but the slope of the angle at the edge of the interlayer, 3.5 in./in., exceeds the shear strain allowables for S-130 at 195°F temperature. The average bond line rupture shear strain for S-130 interlayer at 195°F, low strain rate, is 3.35 in./in. from Reference 18.

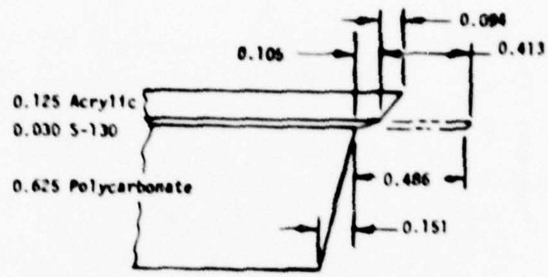


Figure 19. Maximum Temperature Effect on 0.030 Copolymer Interlayer (Sierracin S-130) (Dimensions in Inches).

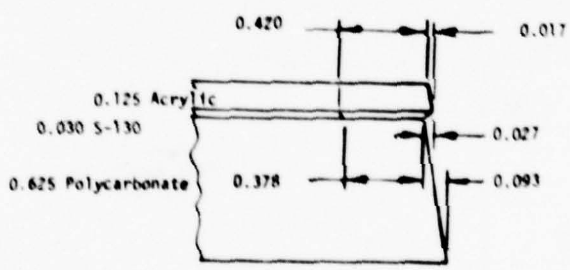


Figure 20. Minimum Temperature Effect on 0.030 Copolymer Interlayer (Sierracin S-130) (Dimensions in Inches).

Edge Frame Clearance

A consideration which must always be made is the total expansion and contraction of the acrylic with respect to the polycarbonate. Figure 15 shows the total excursion to be 0.22 inch for the 0.080-inch interlayer. This is expansion of the acrylic with respect to the polycarbonate external surface. Therefore, the design must allow at least 0.22 inch for expansion over the canopy. This expansion will increase the radius,

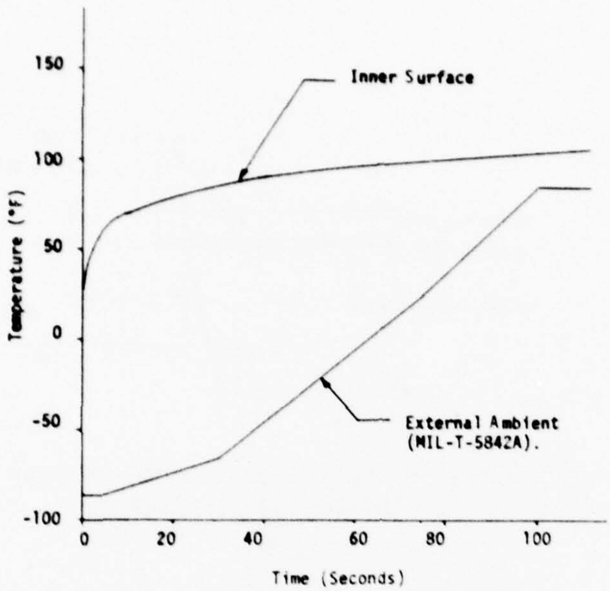


Figure 21. Temperature Profile of Canopy Inner Surface (During Descent with Defog Activated at Top of Descent).

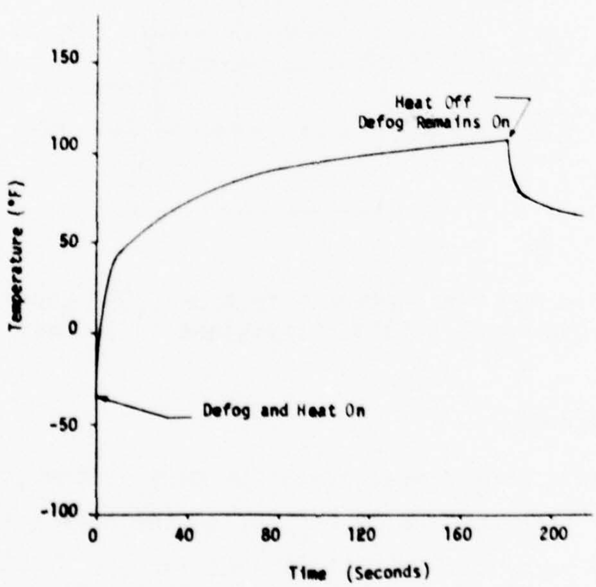


Figure 22. Temperature Profile of Canopy Inner Surface (Following Activation of the Defog System on a Cold Soaked Aircraft at -65°F).

which leads to bolt bending. If the bolt does not bend, the relative expansion of the acrylic will be increased by the distance that the bolt could have bent. The distance the bolt would have bent is 0.139 inch; therefore the total expansion will be approximately 0.22 + 0.139/2 = 0.29-inch. Only half of the amount the bolt would have bent is considered in differential expansion due to the greater expansion and subsequent compression of the outboard surface of the polycarbonate, and tension of the inboard surface. Using the same criteria and Figure 17 for minimum temperature, the contraction will be 0.04 inch for a total excursion of 0.33 inch.

DEFOG SYSTEM PERFORMANCE

No change to the current defog system was proposed for the alternate canopy system design as the current system is adequate to provide defogging whenever it is required. Upon defog activation, most of the air being delivered to the cockpit for air conditioning is diverted to defogging through the 10 slots on either side of the forward canopy. For the first three minutes of operation the air is heated to 158°F and then reverts back to the temperature required by the cockpit. Figure 21 shows the inner surface temperature, with defog activated at the top of descent, as a function of time during descent through the fogging atmosphere as defined in MIL-T-5842A. The canopy inner surface remains above the external ambient temperature throughout the descent. Figure 22 shows the heat up of the inner surface if the defog is activated on an aircraft which has been ground soaked at -65°F. Initial cockpit temperature is assumed at -20°F.

EDGE ATTACHMENT AREA TEMPERATURE STUDY

The edge attachment thermal study was conducted to support the edge joint tests reported in Section VIII. High load concentration and the adverse effects caused by high and low temperatures are justification for critical assessment of materials selections during initial design. The thermal model of the canopy side edge area included the frame and

placement of the attachments because these components affect the transparency temperature. Three thicknesses of polycarbonate were selected for the edge joint tests. The thermal study was conducted on the thinnest section which would be the hottest edge. The laminated section analyzed consisted of 0.625-inch polycarbonate, 0.080-inch acrylic, and an interlayer of 0.080-inch silicone.

Maximum and Minimum Temperature Distribution

The maximum and minimum temperature distribution was calculated at the edge of the transparency and frame area. Figure 23 shows the temperature distribution at the time that the maximum possible temperature occurs at the canopy edge. These maximum temperatures occur at the end of a 10-minute supersonic cruise in a U.S. Standard atmosphere. Figure 24 shows the temperature distribution for the minimum temperatures which will occur at subsonic cruise with a -80°F ram-air temperature. A thicker polycarbonate ply would result in less severe temperatures.

Average Temperature Distribution

In support of the edge attachment test reported in Section VIII; the average polycarbonate temperatures were computed for several flight conditions. The tests were to investigate the edge design reliability when subjected to thermal and load stresses. The following flight conditions were picked to best represent the F-16 flight regime as given in Reference 3; then these same conditions were simulated as described in the subsequent paragraphs.

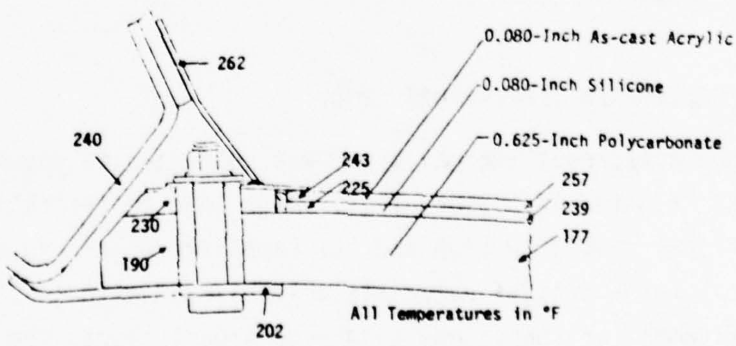


Figure 23. Maximum Temperatures at Canopy Edge.

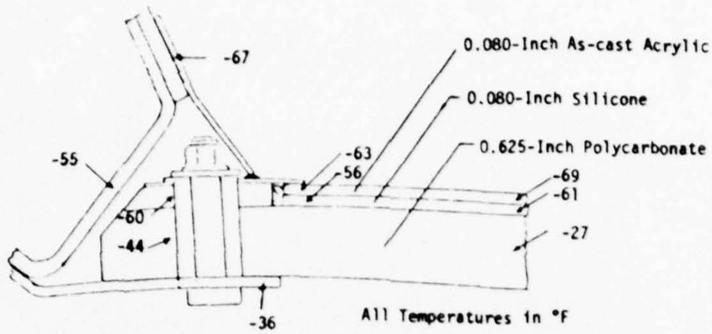


Figure 24. Minimum Temperatures at Canopy Edge.

The simulated flight conditions are as follows:

1. Subsonic cruise - standard atmosphere, 916 pressure cycles
2. Subsonic cruise - hot atmosphere, 102 pressure cycles
3. Supersonic cruise - Mach 1.6 - standard atmosphere, 498 pressure cycles
4. Supersonic cruise - Mach 1.6 - hot atmosphere, 55 pressure cycles
5. Supersonic cruise - Mach 2.0 - standard atmosphere, 219 pressure cycles
6. Supersonic cruise - Mach 2.0 - hot atmosphere, 24 pressure cycles
7. Hot atmosphere ground soak (160°F), followed by a takeoff and climb to a 10-minute Mach 2.2, supersonic cruise in a standard atmosphere - 5 temperature cycles. See Figure 25.
8. Subsonic cruise in a cold atmosphere, followed by a 10-minute supersonic cruise of Mach 2.2 in a standard atmosphere - 5 temperature cycles. See Figure 26.

During the edge member tests, case Number 1 was simulated by maintaining the specimen at -5°F and pressure cycling 916 times. Case Numbers 2

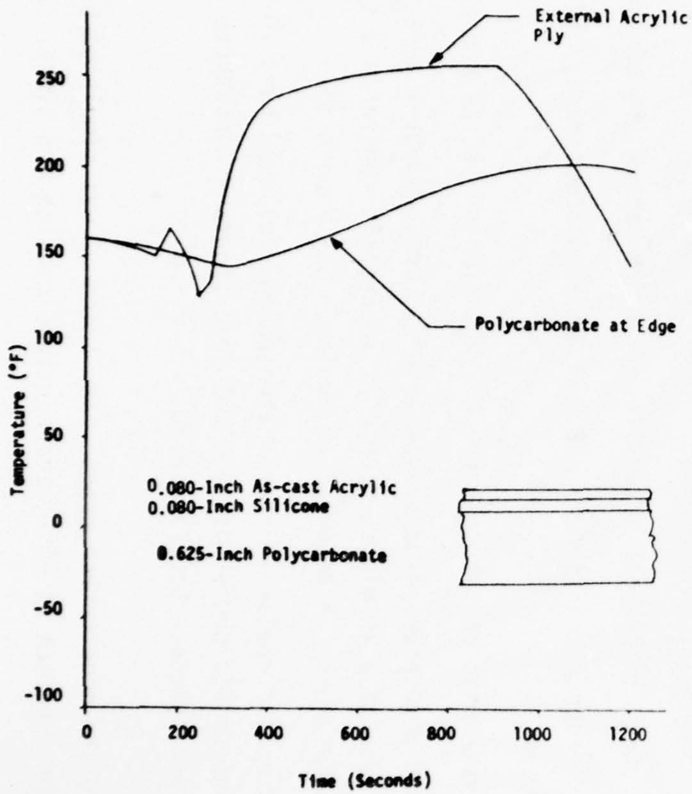


Figure 25. Average Edge Temperatures During Supersonic Cruise (Includes climb from hot ground conditions and descent).

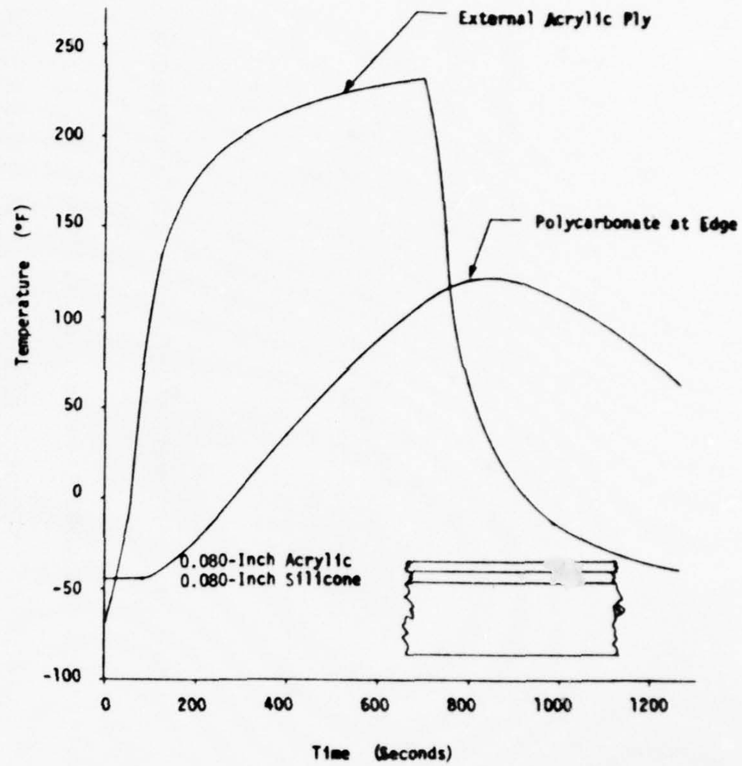


Figure 26. Average Edge Temperatures During Supersonic Cruise (Includes subsonic cruise before and after).

through 5 were simulated by maintaining the specimen at 75°F and pressure cycling a total of 874 times. Case Number 6 was simulated by maintaining the specimen at 160°F and pressure cycling 24 times.

Case Number 7 was simulated by holding a constant pressure and cycling the chamber temperature from -65 to 190°F, 5 times. Case Number 8 was simulated by holding a constant pressure and cycling the chamber temperature from 160 to 245°F, 5 times.

SECTION VI
MATERIALS STUDY

In an endeavor to select materials that could be formed to complex canopy shapes similar to the high performance F-16 aircraft, a series of investigative studies and tests were conducted. The requirements specified in Section II and the thermal criteria established in Section V were utilized to establish the criteria for materials assessments. Within subsequent paragraphs a discussion of these studies and tests are noted, but the detailed results are contained in AFFDL-TR-77-96 (Reference 18).

The determination of the mechanical properties of the materials used in the design of high performance aircraft transparencies involves a knowledge of the load-carrying ability of the transparency materials under low and high strain-rate loading conditions at high and low temperature extremes. The low strain-rate (0.01 in./in./min to 20 in./in./min) mechanical properties of transparency materials are required to design for cabin pressure, thermodynamic loads, aerodynamic loads, and structural carry-through loading. The high strain-rate (100 in./in./min to 12,000 in./in./min) mechanical properties of the transparency materials are required to design for bird strike loading. In the case of low strain-rate loading, the inherent load-carrying ability of the transparency must be such that the limit load can be sustained without excessive deformation or permanent set, requiring a knowledge of the elastic properties of the materials. The load-carrying ability of the windshield under high strain-rate conditions requires a knowledge of the material in the plastic range, where excessive deformation is a design asset for energy absorption in the case of a bird impact shock load. To provide these design mechanical properties, the complete stress-strain histories of candidate materials are required. The critical stress requirements for a windshield were determined to be predominately tensile and compressive in the main load carrying plies, and shear in the interlayer materials.

An investigation of industry test data and publications offered incomplete and/or inadequate information concerning high-performance material properties of many promising aircraft transparency materials. Such properties were considered to be essential to effect a proper balance of structural efficiency, safety, minimum weight and provide a solid basis for design trade-off studies and for establishing material/fabrication specification tests.

Prior industry work in the determination of materials properties was concerned with materials testing at low strain-rate and moderate temperatures. Properties so obtained were found to be average elastic properties without reference to strength variation. Such material properties were considered inaccurate and incomplete for design considerations.

To be of maximum usage, the required material properties should be generated in a test program that considers effects on polycarbonate material due to:

1. Rate of strain at appropriate testing temperature.
2. Effects of fusion bonding and thinning due to forming.
3. Effects of thermal processing history on the material.
4. Effects of in-service aerodynamic heating and service aging.

Also, a sufficient number of tests should always be conducted so that the material properties, allowables, and/or stress-strain curves generated could be presented on a minimum guarantee basis. The tests, as herein outlined, appropriately consider these items in the generation and presentation of the required properties and the exact details are documented in AFFDL-TR-77-96 (Reference 18).

To establish realistic design mechanical properties, a series of material tests were conducted on monolithic and laminated transparent materials as processed by specific fabricators. Design mechanical properties were established for these materials by statistical methods as

described in Reference 18. It is assumed that these design mechanical properties will subsequently be used as established reasonable baseline criteria for the specific materials tested; to be used by design engineers, material suppliers, and transparency fabricators. It is further recommended that initial and periodic testing be accomplished in accordance with transparency specifications and/or material specifications for production windshield/canopies to guarantee that materials meet or exceed these design mechanical properties presented in Reference 18.

MATERIALS AVAILABILITY AND VENDOR CAPABILITIES

Many innovative transparency design concepts have been introduced during this past decade, as well as the development of several new materials. Generally, these transparencies were developed for applications dictating lightweight, aerodynamic shape requirements, or for aesthetic reasons.

The availability of manufacturing state-of-the-art know-how for developing these concepts was limited, the time allotted for development testing was minimal; the results often were very short service life and poor reliability.

For this program, the efforts were directed toward developing transparencies for high-speed low-level-flying aircraft that may develop aerodynamic heat temperatures as high as 285°F.

A determination was made of potentially available materials that could be used in laminated form as shown in Table 5. The raw material suppliers are identified as well as the laminators.

The final selection of materials for the Douglas alternate F-16 transparency design was based on a systematic analysis of a series of tests that were conducted, as well as an assessment of some materials that are currently in flight status on both military and commercial aircraft.

TABLE 5. RAW MATERIAL SUPPLIERS AND LAMINATORS

MATERIAL	VENDOR DESIGNATION	MIL SPEC	INDEX OF REFRACTION	HAZE	OPERATIONAL TEMPERATURE RANGE	THICKNESS AVAIL-ABILITY	RAW MATERIAL SUPPLIERS	LAMINATORS	LIGHT TRANS-MISSION
As-Cast Acrylic	PLEXIGLAS II SWEDLOW TYPE 320	MIL-P-5425	1.49	1%	-65 to 190°F	0.060-0.188	Rohm & Haas Swedlow	PPG Sierracin Swedlow Goodyear	92%
As-Cast modified and partially cross-linked acrylic	PLEXIGLASS 55 SWEDLOW TYPE 350	MIL-P-8184	1.50	1%	-65 to 190°F	0.080-0.188	Rohm & Haas Swedlow	PPG Sierracin Swedlow Goodyear	91%
Stretched Acrylic	STRETCH PLEXIGLASS 55 STRETCH SWEDLOW TYPE 350S	MIL-P-25690	1.499	1%	-65 to 180°F	0.060 - 0.970	Sierracin Goodyear Swedlow	Sierracin Goodyear Swedlow PPG	91%
Poly-carbonate	SL3000	MIL-P-83310	1.586 (avg)	>1%	-65 to 320°F	0.040 - 0.375	General Electric	Texstar Sierracin Swedlow Goodyear PPG	90%
	TUFFAK					0.040 - 0.500	Rohm & Haas		
Cast-In-Place (CIP) Silicone	S-100		1.44	1%	-70 to 350°F		Sierracin	Sierracin	91%
	SS-5272Y		1.409	1%	-70 to 350°F		Swedlow	Swedlow	91%
	FAX-23			0.13%	-65 to 250°F		Goodyear	Goodyear	90.1%
Polyurethane	PPG112		1.4967	0.5%		0.030 per ply	PPG	PPG	85%
	FSX-3A		1.536	1.2%			Goodyear	Goodyear	87.8%
	S-120		1.490	0.3%	-65 to 240°F		Sierracin	Sierracin	86.0%
Copolymer	S-130		1.52				Sierracin	Sierracin	89 - 91.

Table 5 shows the values for light transmission and haze as published by the various suppliers. These values are somewhat academic when related to interlayer materials.

Vendor Capabilities

Douglas has long established a policy that there must be at least two vendors with the capabilities of manufacturing Douglas designed windshield assemblies.

To this end, Douglas tested coupon specimens that were obtained from Texstar, PPG Industries, Sierracin, and Swedlow.

REVIEW OF AVAILABLE MATERIAL PROPERTIES DATA

An exhaustive search was made to gather mechanical properties data for generic usage in the Windshield Technology Demonstrator Program. Literature searches were initiated through the NASA Scientific and Technical Division, the Defense Documentation Center, private and public libraries, the Air Force Flight Dynamics Laboratories, and material suppliers. As the data to perform the required analysis was not available, Douglas began testing transparent materials and was assisted by TerraTek. This testing provided the material properties needed to support the alternate canopy design analysis effort. The data and analysis developed are contained in AFFDL-TR-77-96 (Reference 18).

TESTING

Within the paragraphs that follow are descriptions for a series of tests that were conducted to establish transparency mechanical properties necessary to support the design effort on the alternate canopy designs. The detail test results and analysis are contained in AFFDL-TR-77-96 (Reference 18).

Aerodynamic Heating and Service Aging Effects on Mechanical Properties of Polycarbonate

A series of tests were conducted because it had previously been demonstrated that exposure of bisphenol A polycarbonate materials to temperatures above 80°F (176°F) resulted in accumulative decreases in impact strength, fracture energy, extension to break, and increases in tensile yield strength (Reference 19). Other evidences indicate a degrading of these mechanical properties due to weathering and storage (References 20 and 21). To determine mechanical property changes due to aerodynamic heating, a series of tensile tests were conducted at room temperature on monolithic and fusion bonded polycarbonate specimens after exposure to thermal conditioning representative of the exposure a supersonic aircraft might encounter during its life span. To determine mechanical property changes due to service aging, a series of tensile tests were conducted at room temperature on fusion bonded polycarbonate specimens removed from a four-year-old service-aged canopy.

Low Strain Rate Tensile Mechanical Properties Testing of Monolithic Polycarbonate Materials

A series of tests were accomplished to establish complete stress-strain curve design tensile mechanical properties for monolithic and fusion bonded polycarbonate materials, processed by four manufacturers, at various temperature conditions and at low strain-rates. Test specimens were made from bird impact tested canopies, or furnished by windshield/canopy manufacturing processors. The primary reason for these tests was to provide actual and design allowables for computer analysis development and future designs as described in Section V. Additional uses for the test results were to provide for evaluation of materials and processors, to conduct trade-off design studies, windshield static load design analysis, and to provide test criteria for windshield/canopy design specification control documents. Tests were conducted per ASTM standard methods. Maximum and minimum test temperatures were established for tests based on the flight profile of supersonic aircraft.

Low Strain Rate Shear Mechanical Properties Testing of Laminated Interlayer Materials

A series of tests were conducted to establish complete stress-strain curve shear mechanical properties of interlayer materials as processed by specific windshield/canopy fabricators. The primary reason for these tests was to provide actual and design allowables of processed interlayer materials for computer analysis development and future design. Additional uses for the test results were to provide for evaluation of materials and processors, to conduct trade-off design studies, canopy static load analysis, and to provide test criteria for the windshield/canopy design specification control documents. Test specimens were made from bird impacted test canopies (Section VII), or furnished by specific windshield/canopy fabricators. Two types of shear tests were conducted, a generally used compression double shear test (ASTM standard being prepared) and a unique torsional shear test. Results of these two prepared types of testing were compared to arrive at the most accurate means of testing. Maximum and minimum test temperatures were established for tests based on the flight profile of a supersonic aircraft.

High Strain Rate Tensile Mechanical Properties Testing of Monolithic Polycarbonate Materials

A series of tests were conducted to provide complete stress-strain design tensile mechanical properties, at high strain rates, of monolithic polycarbonate materials as processed by specific windshield/canopy fabricators. The primary use for these mechanical properties was for computer analysis development and future design. Additional uses were to provide for evaluation of materials and processors and to conduct design trade-off studies. Tests were conducted at the maximum and minimum temperature conditions possible for a supersonic aircraft windshield or canopy at the time of a bird strike. Maximum strain rates due to a bird strike were determined from bird impact tests on actual full-size windshields and canopies.

High Strain Rate Shear Mechanical Properties Testing of Laminated Interlayer Materials

A series of tests were conducted to provide stress-strain design shear mechanical properties of laminated interlayer materials as processed by specific windshield/canopy fabricators at high strain rates. The primary use for these mechanical properties was for computer analysis development and future design. Additional uses were to provide for evaluation of materials and processors. Tests were conducted at maximum and minimum temperature conditions possible for a supersonic aircraft windshield or canopy during a bird strike. Maximum strain rates were determined from bird impact tests of actual full-size windshields and canopies.

SECTION VII
BIRD IMPACT TESTS RESULTS AND APPLICATIONS

Bird impact hazards to high speed, low flying aircraft have become one of the major flight safety problems. For the F-16 aircraft early bird impact development tests had been conducted on 0.375-inch and 0.500-inch thick uncoated monolithic polycarbonate canopies at speeds of 350 knots utilizing four pound birds. The early tests were conducted at room temperature with the birds impacting the canopies at eye level on the centerline of the canopies. At impact one of the canopies deflected approximately seven inches and caused extensive damage to an anthropomorphic head that was installed to represent a pilot's head. The extent of injury a pilot would have received under such impact conditions is unknown, but it is assumed to be critical. Based on the results of these development tests, a 0.500-inch thickness was selected for the initial production canopy.

A subsequent production test program was designed and conducted on 0.500-inch and 0.625-inch thick coated monolithic polycarbonate. For this series of tests there were five static loading/dynamic unloading tests and thirty bird strike tests conducted at hot, cold and ambient temperatures. Each canopy was instrumented with thermocouples and selected canopies were instrumented with strain gages so that specific data could be documented during testing for subsequent assessment regarding:

- Deflection - canopy to crew head clearance at impact.
- Deflection - canopy to HUD (Head-Up-Display) clearance at impact.
- The additional protection provided by an increase in transparency thickness.
- To establish the upper limits for impacts with 2 and 3 pound birds.

- Comparison between impacts with 2, 3 and 4 pound birds.
- Temperature effect on bird impact resistance of polycarbonate.
- Canopy sensitivity to bird impact location.
- Comparison of static and bird impact strain readings.
- To provide data that could be utilized to compare monolithic versus laminated canopy designs.
- To compile a data bank for potential correlation to bird impact analytical methods.

The production test program was optimistically developed, as shown in the Appendix, from reported development test results. The initial selection of a four-pound bird projected at a coated production canopy at a velocity of 350 knots proved to be grossly in error. Consequently, the testing could not be conducted as planned and the velocities had to be greatly reduced.

This section presents the results of these static loading/dynamic unloading and bird impact tests, an analysis of the effects of the protective coatings on bird impact, design applications, and conclusions.

TEST RESULTS

The test plan is contained in the appendix of this report. Failure of the first test canopy when impacted by a four-pound bird traveling at 349 knots resulted in a revision to certain test objectives and test parameters. Test velocities were reduced to a level determined to be consistent with canopy survival; deflection curves and strain data were documented at severely reduced velocities; a limited effort was directed toward impact location sensitivity; and the benefits of a thicker transparency were investigated, but impact resistance to smaller birds was not developed.

Table 6 lists the tests conducted and test results. A total of 30 bird impact tests and five static tests were conducted on eight coated

TABLE 6. STATIC/BIRD IMPACT TEST RESULTS

TEST NO.	TRANS-PARENCY NO.	NOMINAL THICK-NESS (IN.)	TARGET POINT	TARGET TEMP. (°F.)	AVERAGE TEMP. (°F.) OUTSIDE	INSIDE	STRAIN GAGES	V (KNOTS)	LOAD/ BIRD WGT. (LBS.)	REMARKS
001	C1/0030	0.50	A	75	70.4	65.9	Yes	Static	2200	
001A			A	75			Yes	Static	2200	
002			A	195	195.6	186.3	Yes	Static	2200	
003			A	-35	-77.7	-47.3	Yes	Static	2200	
004			A	75	61.9	65.8	Yes	349	4.05	Failed Sta. 102 to 116
006	C4/0029	0.50	A	75	73.8	83.7	No	344	2.16	Failed Sta. 144 to 170
011	C2/0040	0.50	A	75	75.8	73.4	No	176	4.07	Failed Sta. 144 to 167
012	C5/0027	0.50	A	75	72.5	72.4	No	191	2.10	
013			A	75	74.3	76.2	No	124	4.06	
013A			A	75	66.1	74.9	No	140	4.01	
014A			A	195	189.9	200.2	No	159	4.14	
015A			A	-35	-30.4	-26.4	No	165	4.04	
015B			A	195	207.1	200.1	No	173	4.10	
015C			A	-35	-18.1	-21.1	No	173	4.09	
015D			C	75	80.6	80.5	Yes	155	4.01	
015E			A	75	84.3	85.0	Yes	174	4.18	Failed Sta. 165 to 170
018	C3/0043	0.50	A	75	75.1	74.0	Yes	149	4.07	
019			A	195	182.4	200.1	Yes	151	4.09	
020			A	-35	-41.9	-38.5	Yes	148	4.00	
020A			A	75	79.1	82.9	Yes	158	4.18	Crack-Aft End
020B			A	75	83	84	Yes	173	4.18	Failed Sta. 145 to 157
021	C6/0001	0.62	A	75	83.0	71.8	No	Static	2500	
028			A	75	85.4	86.0	No	155	4.02	
023			A	75	81.9	83.3	No	188	4.05	
024			A	75	80.8	81.9	No	253	4.20	Failed Sta. 133 to 165
022	C8/0004	0.62	A	75	75.4	86.8	Yes	202	4.10	
022A			A	195	193.4	208.3	Yes	200	4.01	
022B			A	-35	-39.5	-40.7	Yes	199	4.09	
022C			A	75	76.2	76.8	Yes	201	4.01	HUD Installed
022D			A	75	75.1	73.3	Yes	230	4.01	
022E			B	75	72.9	76.0	Yes	231	4.12	
022F			A	75	77.1	78.9	Yes	363	4.02	Failed Sta. 136 to 176
028A	C7/0005	0.62	B	75	80.5	80.4	Yes	233	4.07	Failed-Cracks at Sta. 136 & 166
GD1	Uncoated	0.50	A	75			No	344	2.16	Conducted by General Dynamics
GD2			A	75			No	351	3.12	Conducted by General Dynamics
GD3			A	75			No	340	4.01	Conducted by General Dynamics
GD4			A	75			Yes	158	4.07	
GD5			A	75	62.0	68.2	Yes	362	4.06	Failed Sta. 136 to 176

transparencies and one uncoated transparency. Three additional bird impact tests were conducted by General Dynamics on the uncoated transparency. The temperatures are the average of four thermocouples and were taken near the point of impact. The velocities and bird weights are measured values. Target point A is centerline high, B is centerline low and C is off-center high, as shown in Figure 27.

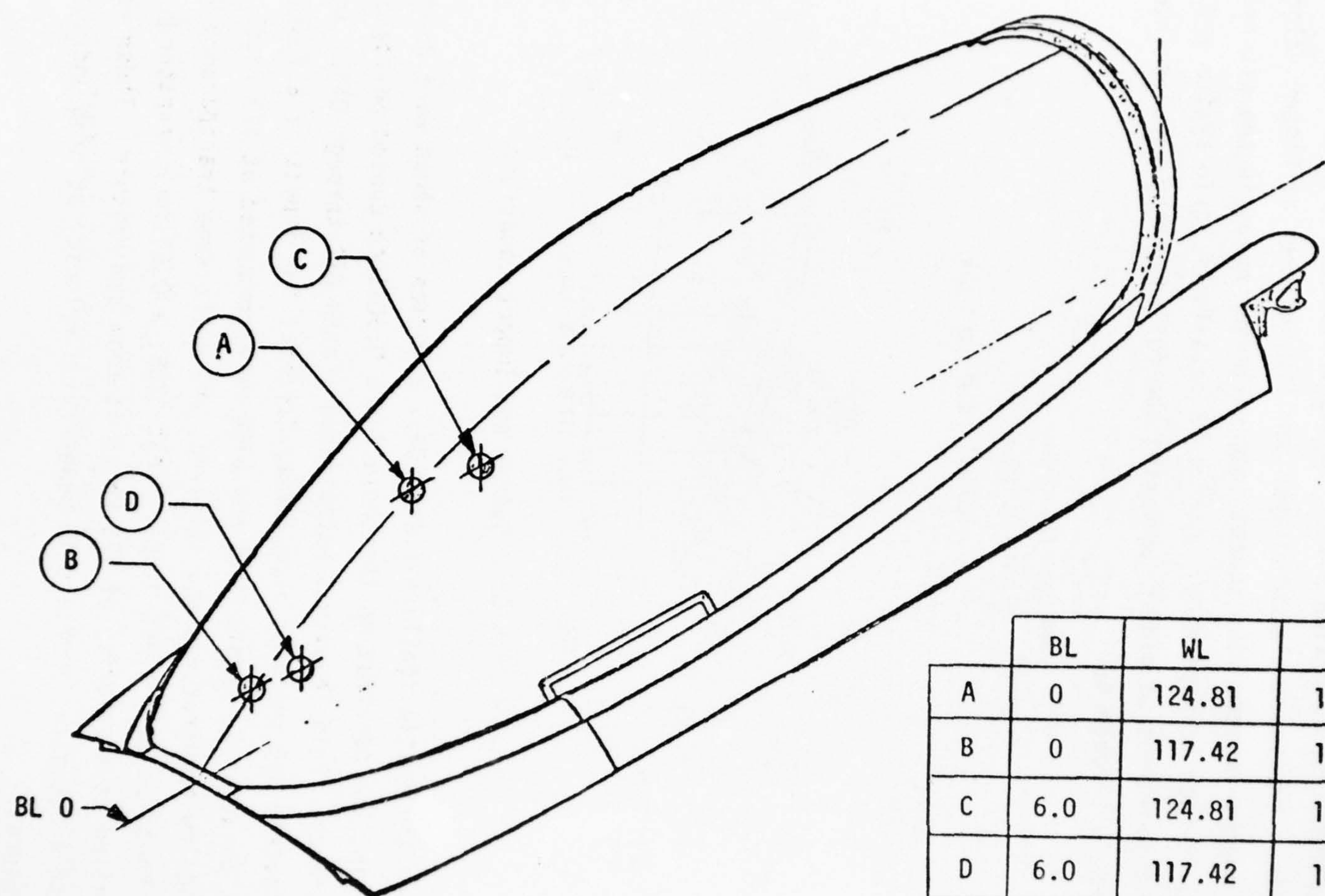
The minimum thicknesses documented in Table 7 were measured on the centerline, generally at Station 138.5. No measurements were documented aft of this location.

Eight transparencies furnished by the Air Force and one transparency provided by General Dynamics for this test series were destroyed by the bird impact tests.

TABLE 7. TRANSPARENCY THICKNESS MEASUREMENTS

TRANSPARENCY NUMBER	NOMINAL THICKNESS (INCHES)	MINIMUM THICKNESS (INCHES)	PERCENT REDUCTION (%)
C1/0030	0.50	0.412	18
C2/0040	↓	0.417	17
C3/0043	↓	0.434	13
C4/0029	↓	0.401	20
C5/0027	↓	0.417	17
C6/0001	0.625	0.523	16
C7/0005	↓	0.523	16
C8/0004	↓	0.553	11

Comparative evaluations of the various specimens were made utilizing the deflection curves and strain readings, and an assessment was made of the capability of each transparency to prevent bird penetration.



	BL	WL	FS
A	0	124.81	112.50
B	0	117.42	100.00
C	6.0	124.81	115.16
D	6.0	117.42	102.75

Figure 27. Location of Bird Impact Coordinates.

The first failure (Test 004) occurred on a 0.50-inch transparency at the point of impact (Location A). The impact speeds were reduced and subsequent shots (Tests 006 and 011) produced failures to the aft end of the transparencies, approximately 35 inches aft of the point of impact. The 0.62-inch transparencies displayed similar failures at higher velocities. The high energy impacts (350 knots) induced larger holes in the aft half of each 0.62-inch transparency. Figure 28 details the grid system and the impact locations to aid in comparing the failed transparencies shown in Figures 29 through 42.

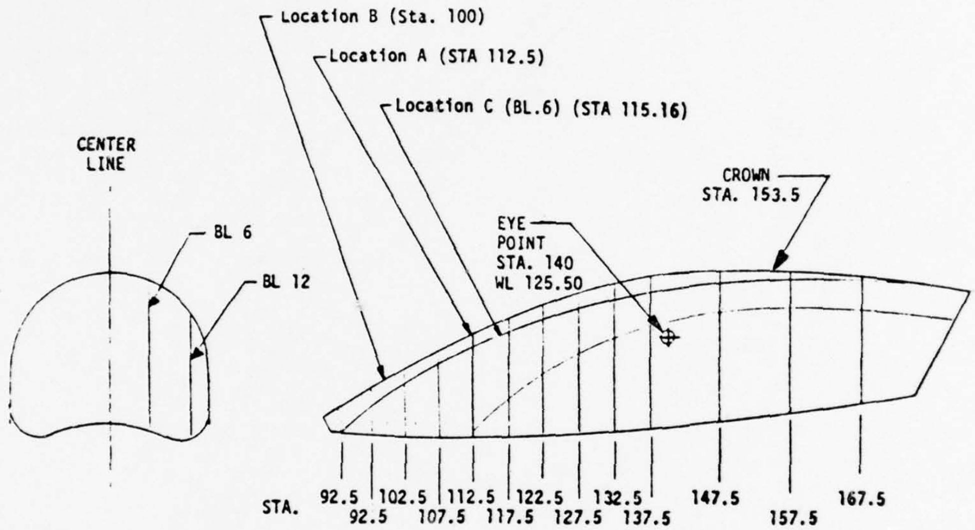


Figure 28. Grid System and Impact Locations.

After the initial failure (Test 004), a series of shots were conducted to pinpoint the penetration threshold for a 0.50-inch coated polycarbonate transparency. This level was established (Tests 012 through 015A) as approximately 165 knots at room temperature for the impact of a four-pound bird at Location A. Tests 015B and 015C were conducted at 173 knots at high and low temperatures and survived, while the same transparency failed at 174 knots at 75°F (Test 015E). Test 020B, a 0.50-inch transparency, also failed at a velocity of 173 knots at room temperature. These results indicate a potential increase in penetration velocity at high and low temperatures.

Strain gages were added to Transparency C5/0027 and Test 015D was conducted to investigate location sensitivity. This off-center/high impact produced insignificant deflections, thus implying reduced impact sensitivity at locations away from the symmetrical centerline.

Transparency C3/0043 was instrumented and tested to gather strain data for potential application to the development of analytical methods for bird impact.

Test 020A was conducted to ascertain the effect of multiple shots on a canopy. The deflection curve of Test 020A was compared to the deflection curve of Test 018. The deflection was less for Test 020A at a slightly higher velocity than Test 018. Test 020B was performed to explore failure repeatability by comparison with Test 015E. The velocities and bird weights were identical. Both transparencies failed. Figures 35 and 36 display the differences in the failure.

Tests on the coated 0.62-inch transparencies established that the penetration velocity was between 188 and 253 knots for a 4.0 pound bird impact at location A and 75°F. A velocity of 200 knots was picked for subsequent tests.

A series of seven tests were conducted on a 0.62-inch transparency, C8/0004 (Tests 022 through 022F). Strain data and deflection curves were recorded. The HUD assembly was installed for Test 022C and the deflection was reduced by the HUD assembly. The A and B impact locations were compared and the maximum deflection was greater for the impact at B (Tests 022D and Test 022E). Hot and cold tests were compared. Frost on the transparency prevented deflection measurements on the low temperature test (Test 022B). The maximum deflection for the high temperature test (Test 022A) is greater than the maximum deflection for the test conducted at 75°F.

Test 028A was conducted on a new canopy to gather strain data in the area of the impact at Point B. The transparency was impacted at 233 knots

and failed. A similar test (022E) did not fail a 0.62-inch transparency at 231 knots. The temperatures and bird weights were very close. These test results indicate that the penetration velocity threshold is approximately 231/233 knots for impacts at Location B.

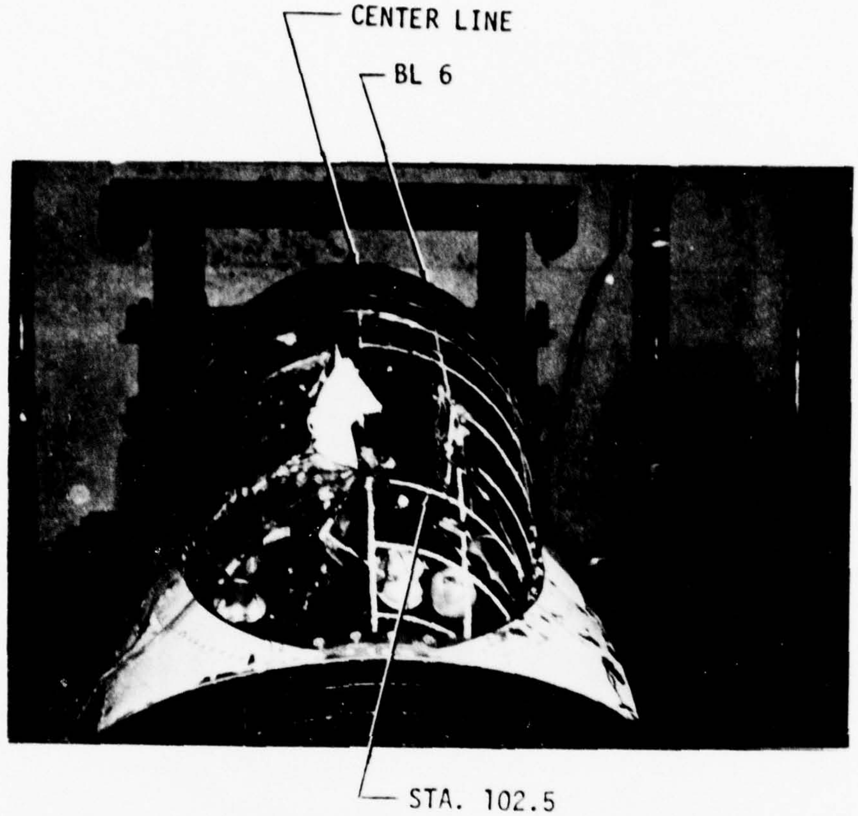
Three tests (GD1, GD2 and GD3) were conducted by General Dynamics (GD) personnel to a GD test plan on an uncoated 0.50-inch transparency. Two-, three- and four-pound bird shots were completed at 350 knots without failure.

This same transparency was tested per the Douglas test plan (Appendix) to compare the deflection of an uncoated transparency to a coated transparency. A comparison of Test GD4 to Test 018 indicates twice as much maximum deflection for the uncoated transparency as for the coated transparency. The results of this test indicate that the protective coating on the polycarbonate tends to increase the material stiffness.

Test GD5 unexpectedly failed the transparency. It had been previously impacted at the same velocity (GD3) and survived. This test did provide strain data at the higher velocity.

Many of the bird impacted transparencies displayed large concentrations of cracks in the protective coatings on the inner surface. These cracks were oriented in a transverse direction, symmetrical about the centerline and located from Station 130 and aft. These cracks are visible in Figures 34, 35 and 36.

Strain rates were calculated from the strain/time curves for bird impact. The values varied from 300 to 4200 in./in./min.



Test Number	004
Velocity:	349 Knots
Bird:	4.05 lbs
Temperature:	61.9/65.8°F
Thickness:	0.50-in. nominal
	0.412 minimum
Hole Size:	Sta. 102.5 to 116, 12 in. wide

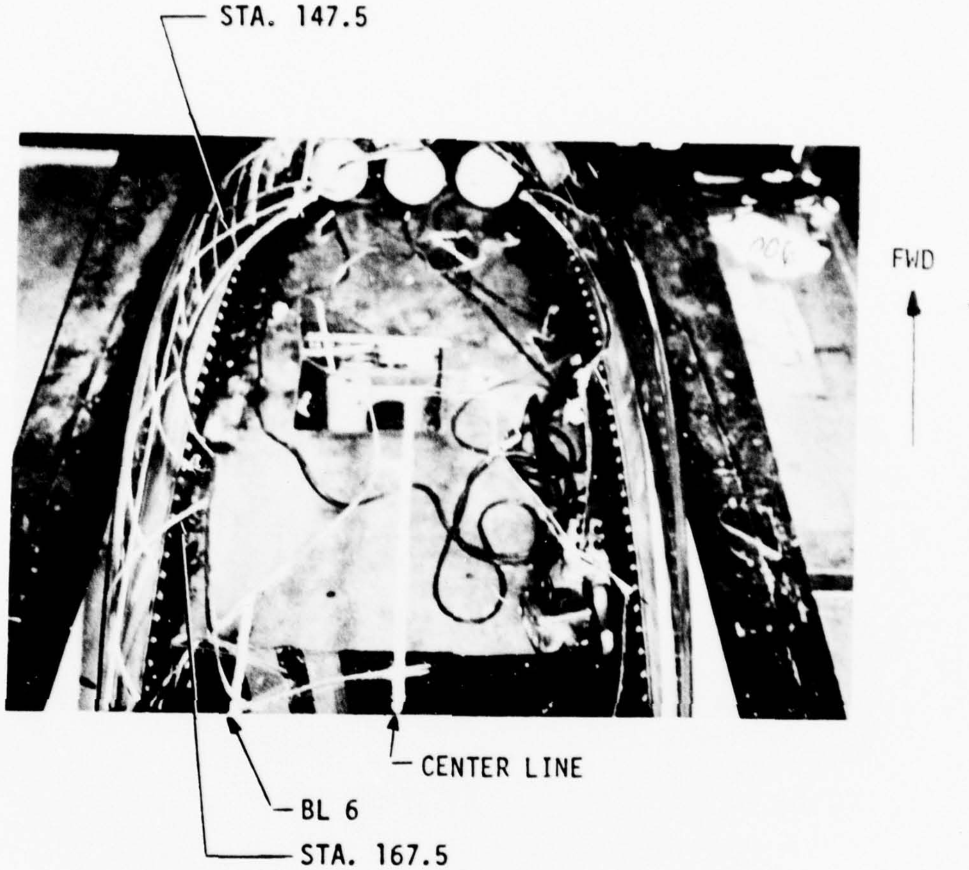
Figure 29. Post-Test Condition of Transparency Number C1/0030 (looking aft).

BL 6

STA. 112.5

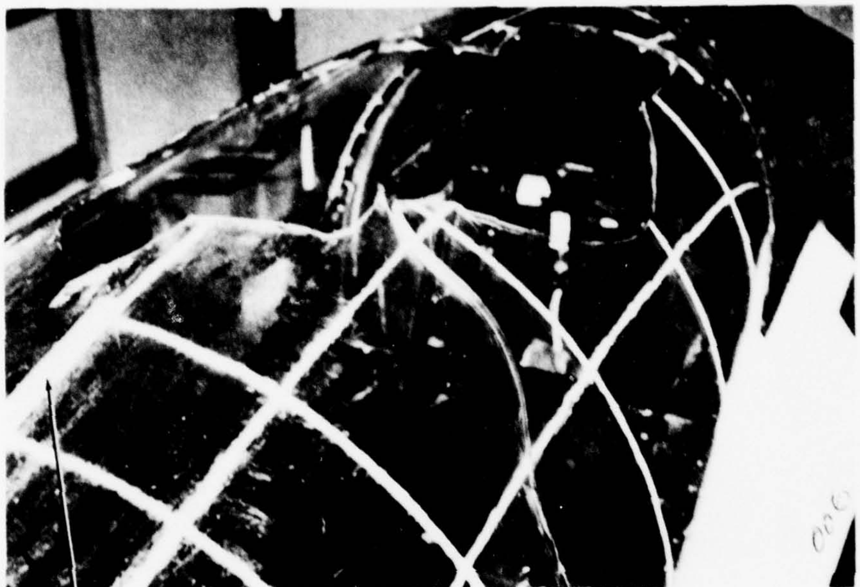
Test Number	004
Velocity:	349 knots
Bird:	4.05 lbs
Temperature:	61.9/65.8°F
Thickness:	0.50-in. nominal
Hole Size:	Sta. 102.5 to 116, 12 in. wide

Figure 30. Post-Test Condition of Transparency Number C1/0030.



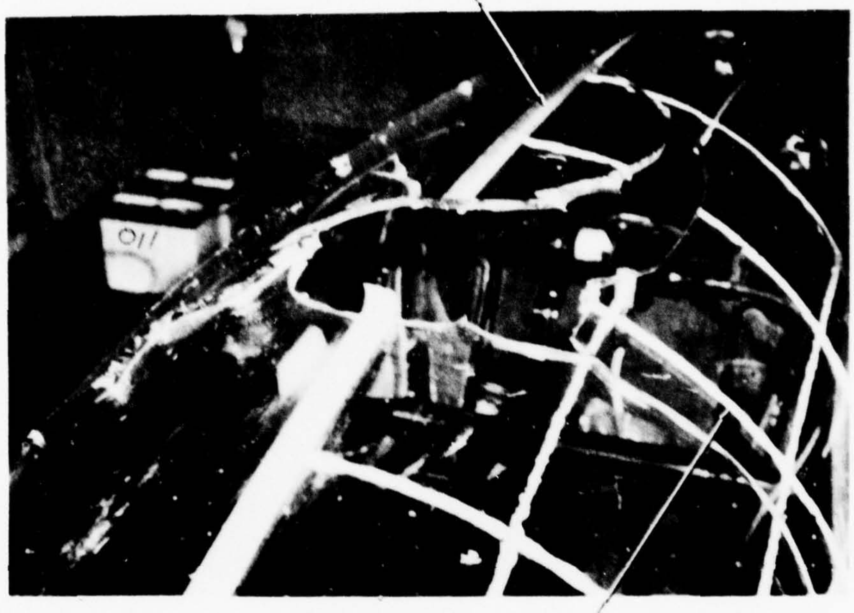
Test Number	006
Velocity:	344 knots
Bird:	2.16 lbs
Temperature:	73.8/83.7°F
Thickness:	0.50-in. nominal
	0.401-in. minimum
Hole Size:	Sta. 144 to 170, 20 in. wide

Figure 31. Post-Test Condition of Transparency Number C4/0029 (looking down).



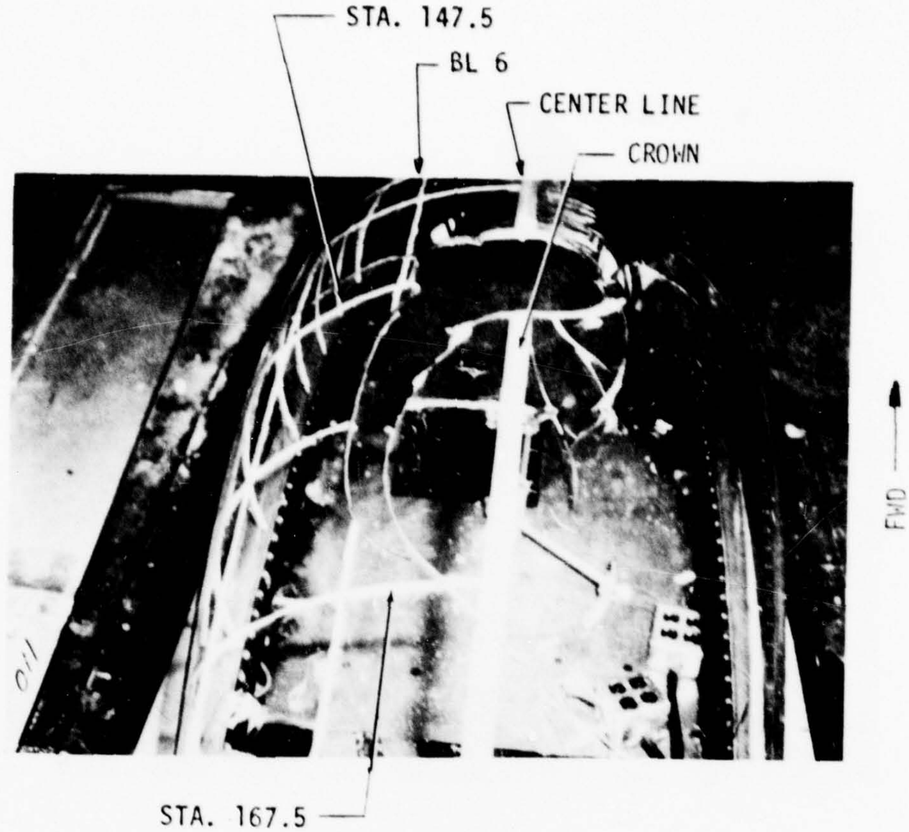
Test Number	006
Velocity:	344 knots
Bird:	2.16 lbs
Temperature:	73.8/83.7°F
Thickness:	0.50-in. nominal
	0.401-in. minimum
Hole Size:	Sta. 144 to 170, 20 in. wide

Figure 32. Post-Test Condition of Transparency Number C4/0029 (looking aft).



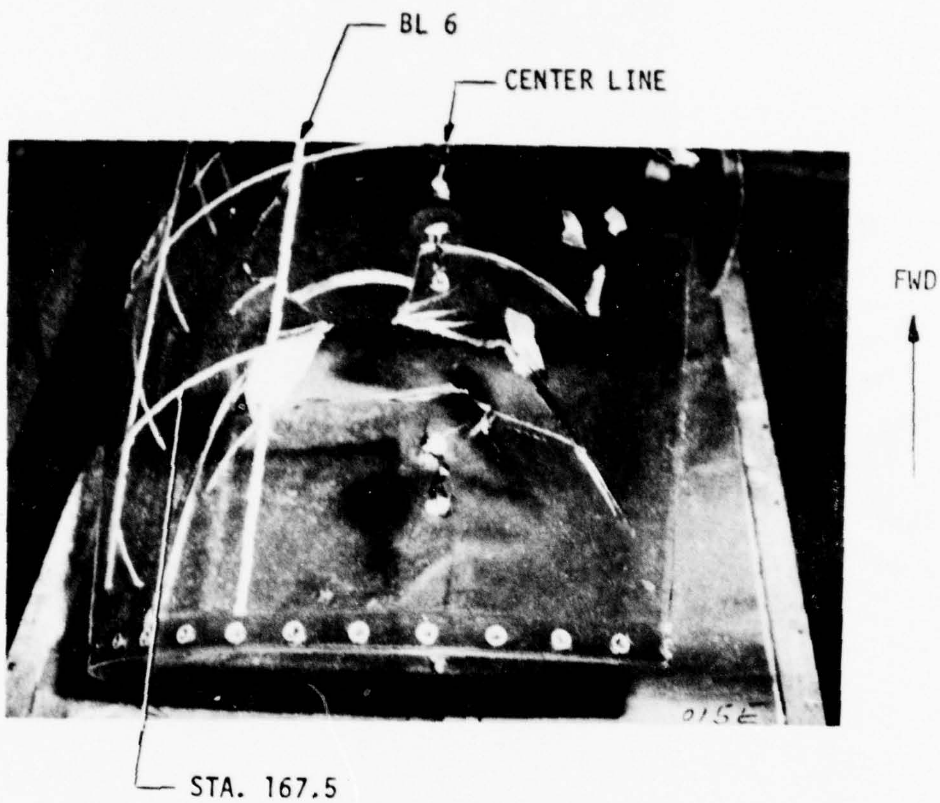
Test Number	011
Velocity:	176 knots
Bird:	4.07 lbs
Temperature:	74.8/73.4°F
Thickness:	0.50-in. nominal
	0.417-in. minimum
Hole Size:	Sta. 144 to 167, 12 in. wide

Figure 33. Post-Test Condition of Transparency Number C2/0040 (looking aft).



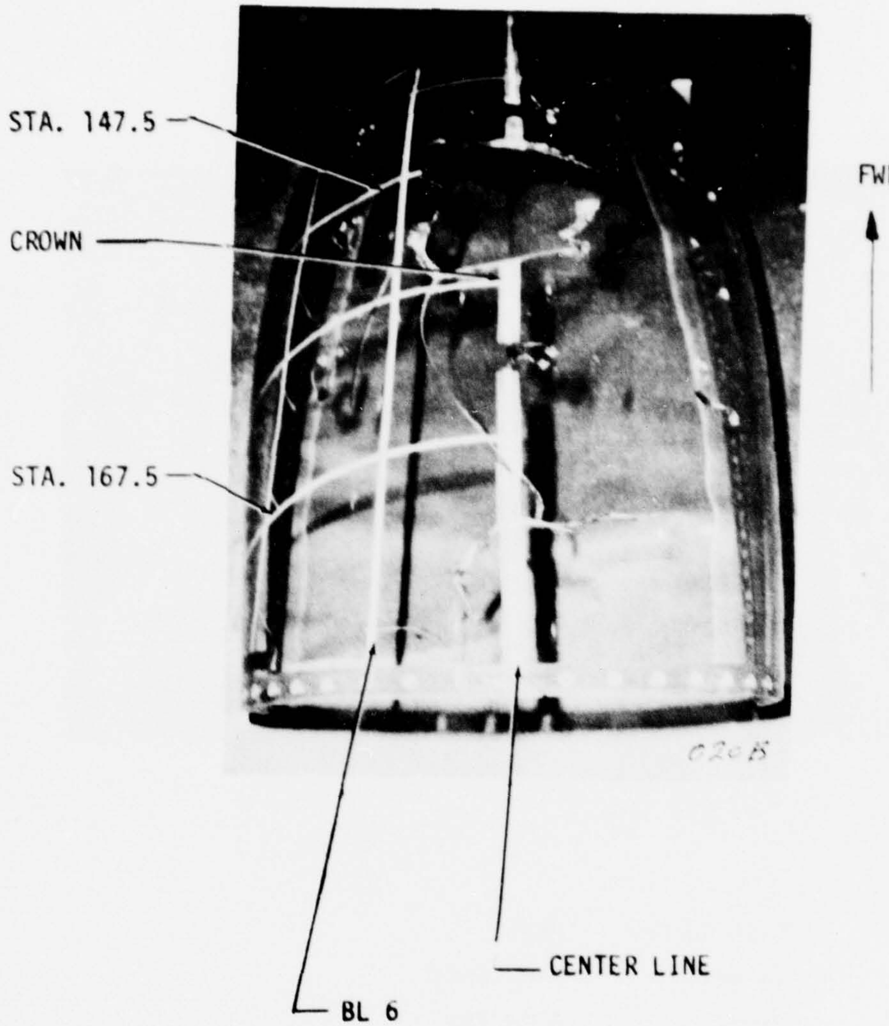
Test Number	011
Velocity:	176 knots
Bird:	4.07 lbs
Temperature:	75.8/73.4°F
Thickness:	0.50-in. nominal
	0.417-in. minimum
Hole Size:	Sta. 144 to 167, 12 in. wide

Figure 34. Post-Test Condition of Transparency Number C2/0040 (looking down).



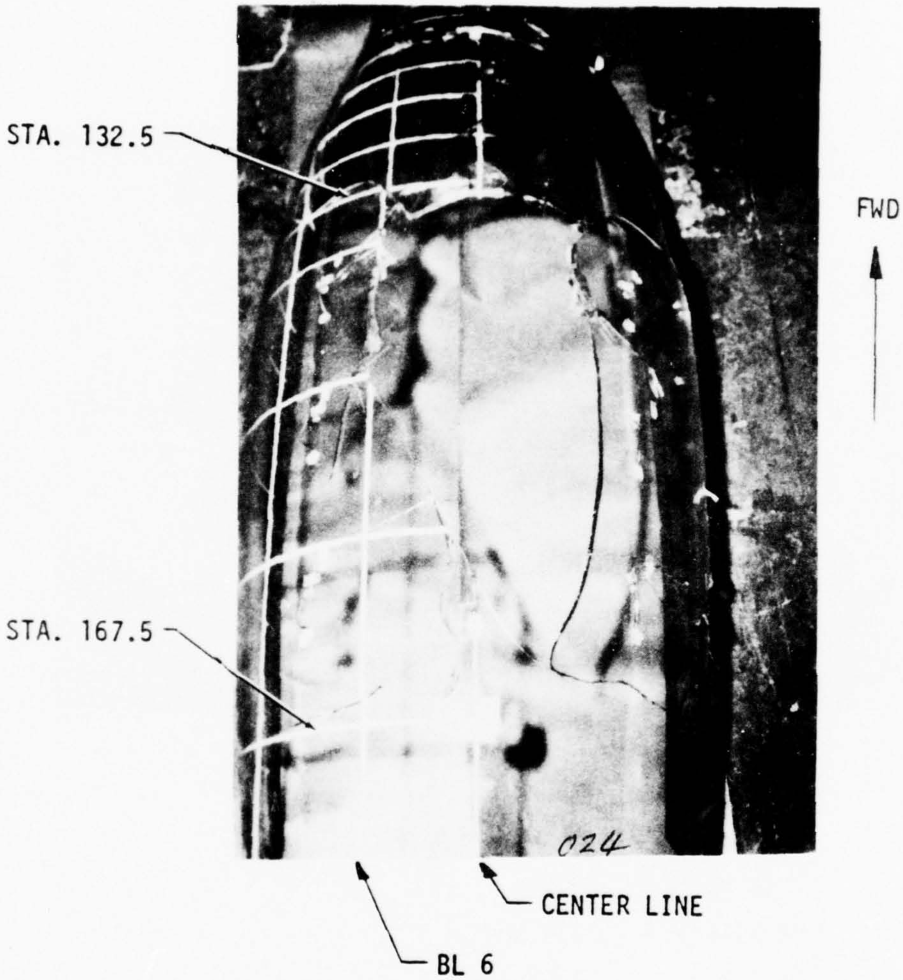
Test Number	015E
Velocity:	174 knots
Bird:	4.18 lbs
Temperature:	84.3/85.0°F
Thickness:	0.50-in. nominal
	0.417-in. minimum
Hole Size:	Sta. 165 to 170, 12 in. wide

Figure 35. Post-Test Condition of Transparency Number C5/0027 (looking down).



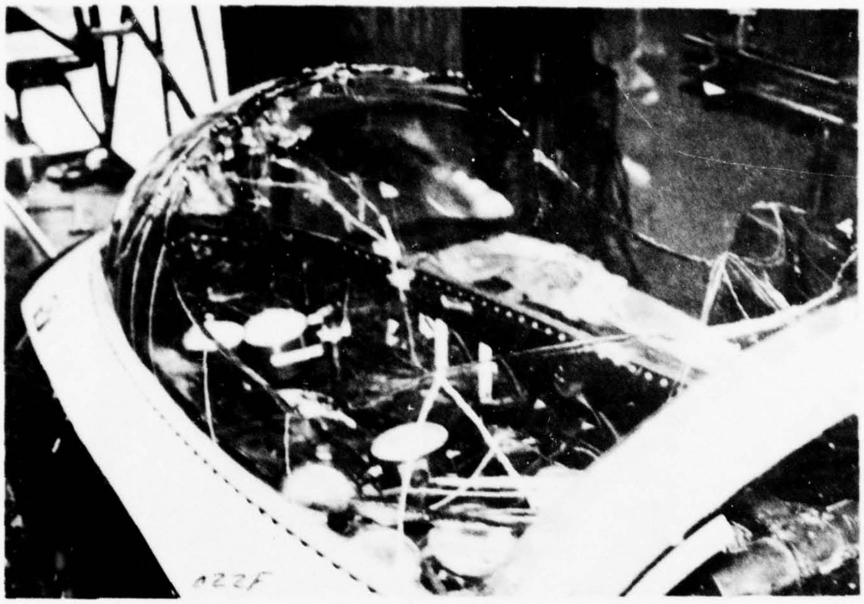
Test Number	020B
Velocity:	173 knots
Bird:	4.18 lbs
Temperature:	83/84°F
Thickness:	0.50-in. nominal
	0.434-in. minimum
Hole Size:	Sta. 145 to 147, 11 in. wide

Figure 36. Post-Test Condition of Transparency Number C3/0043 (looking down).



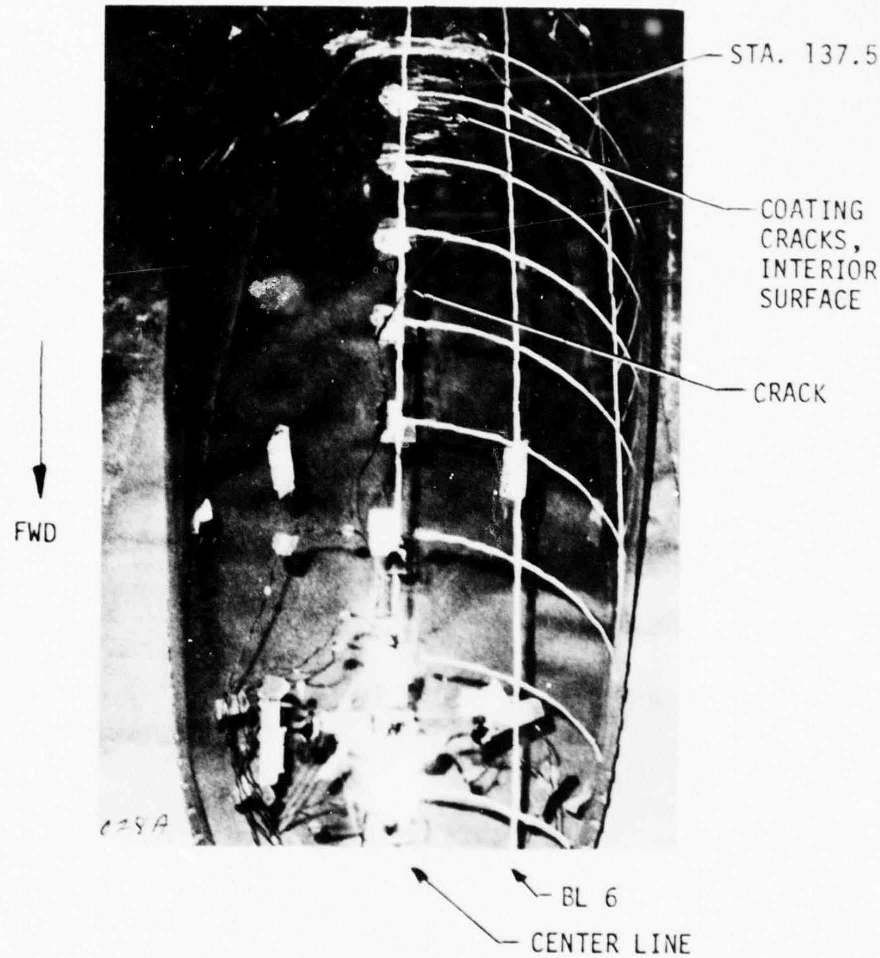
Test Number	024
Velocity:	253 knots
Bird:	4.2 lbs
Temperature:	80.8/81.9°F
Thickness:	0.62-in. nominal
	0.523-in. minimum
Hole Size:	Sta. 133 to 165, 12 in. wide

Figure 37. Post-Test Condition of Transparency Number C6/0001 (looking down).



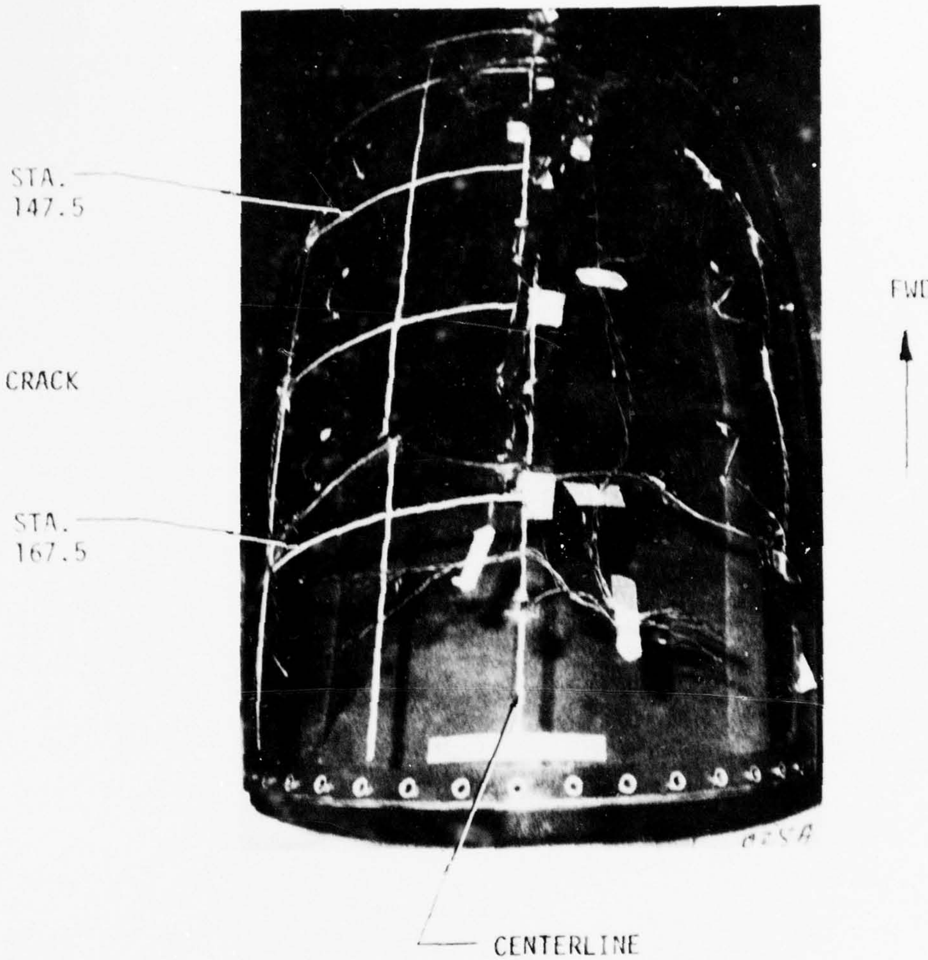
Test Number	022F
Velocity:	363 knots
Bird:	4.02 lbs
Temperature:	77.1/78.9°F
Thickness:	0.62-in. nominal
	0.523-in. minimum
Hole Size:	Sta. 136 to 176, 40 in. wide

Figure 38. Post-Test Condition of Transparency Number C8/0004 (looking forward).



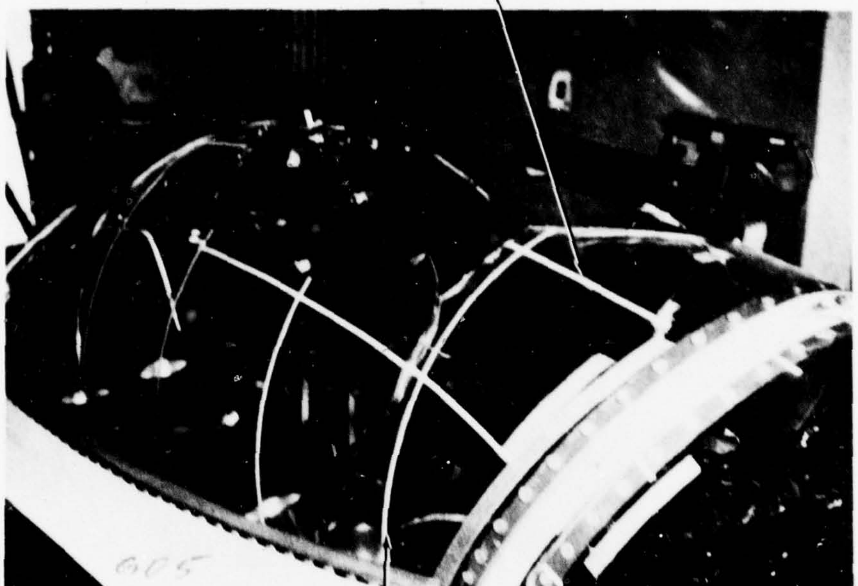
Test Number	028A
Velocity:	233 knots
Bird:	4.07 lbs
Temperature:	80.5/80.4°F
Thickness	0.62-in. nominal
	0.523-in. minimum
Hole Size:	Transverse cracks at Sta. 136 and 166 Crack along center line, Sta. 102 to 136

Figure 39. Post-Test Condition of Transparency Number C7/0005 (looking down forward end).



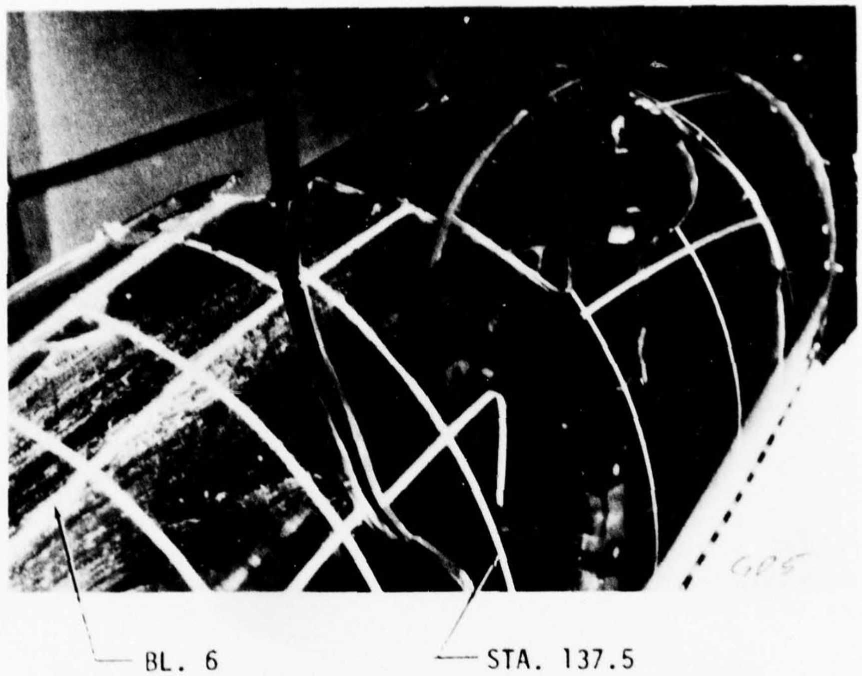
Test Number	028A
Velocity:	233 knots
Bird:	4.07 lbs
Temperature:	80.5/80.4°F
Thickness:	0.62-in. nominal
	0.523-in. minimum
Hole Size:	Cracks at Sta. 136 and 166

Figure 40. Post-Test Condition of Transparency Number C7/0005 (looking down aft end).



Test Number	GD5
Velocity	362 knots
Bird:	4.06 lbs
Temperature:	62.0/68.2°F
Thickness:	0.50-in. nominal
Hole Size:	Sta. 140 aft, 24 in. wide

Figure 41. Post-Test Condition of Uncoated Transparency (looking forward).



Test Number	GD5
Velocity:	362 knots
Bird:	4.06 lbs
Temperature:	62.0/68.2°F
Thickness:	0.50-in. nominal
Hole Size:	Sta. 140 aft, 24 in. wide

Figure 42. Post-test Condition of Uncoated Transparency (looking aft).

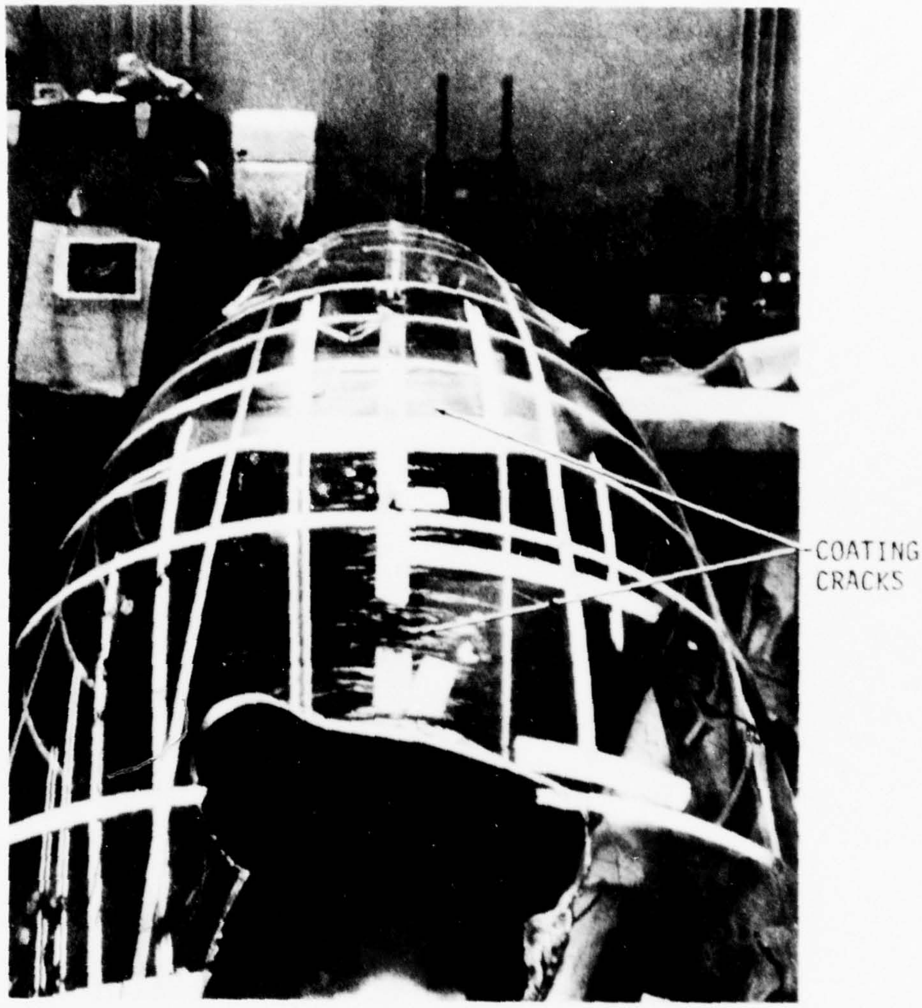
PROTECTIVE COATING ANALYSIS

The unexpected failure of Transparency C1/0030 (Test 004) and subsequent failures precipitates an investigation to determine the relationship of the polycarbonate coating to the bird penetration failure.

Transparency Number C3/0043 was subjected to visual and microscopic examination. The aft portion of the transparency, Figure 43, showed a series of coating cracks perpendicular to the centerline. A light source was oriented 45 degrees to the apex of the transparency in order to view and photograph the coating cracks and fissures. The coating cracks were predominately on the interior surface which was in tension during bird impact.

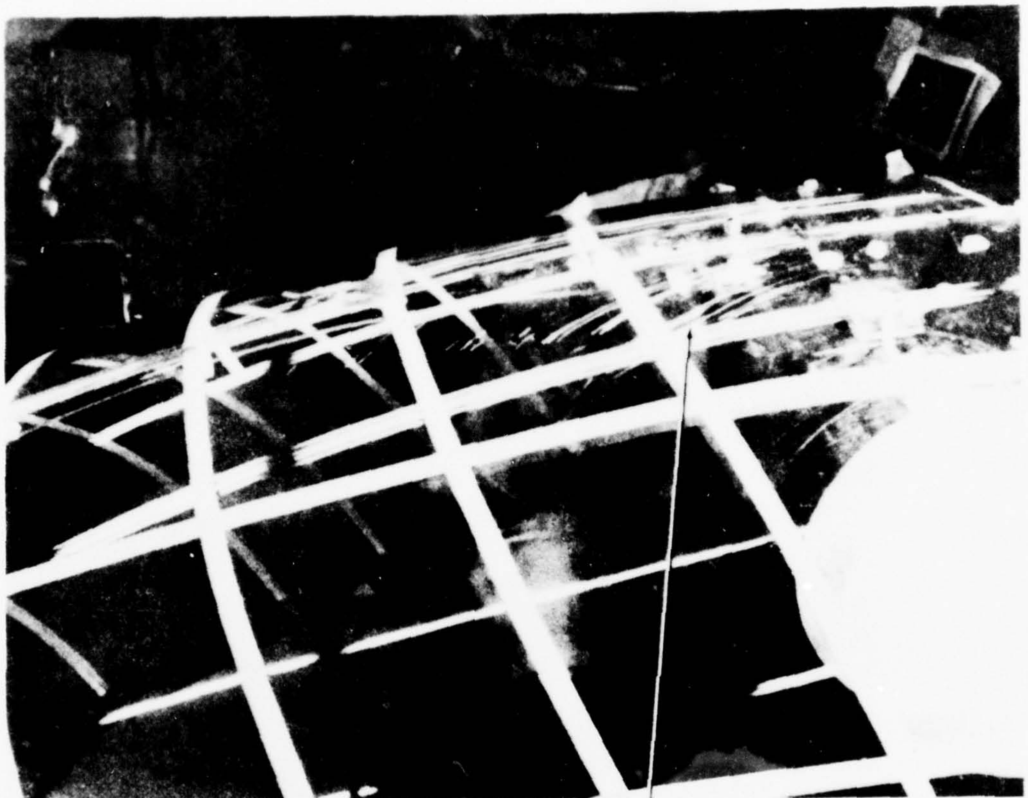
The forward section of Transparency C3/0043, Figure 44, showed a series of coating cracks parallel to the centerline in the bird impact area. The cracks continued aft and changed orientation from parallel to perpendicular with respect to the transparency centerline. An explanation of this change in direction is that the deflection footprint was approximately circular at the impact area and then generated into a transverse deflection wave. The deflection wave aft of the impact area, oriented perpendicular to the transparency centerline, traveled from the impact area to the aft canopy bow frame.

One of the ruptured pieces from bird impacted canopy C3/0043, Figure 45, was examined because of significant secondary cracks. Visual and low powered microscopic examinations revealed a series of fine hairline coating cracks oriented at various angles to the canopy centerline and to the cracked polycarbonate surfaces. One ruptured section had an isolated crack fissure approximately 1.00-inch long and about 0.2-inch deep; another isolated crack was about 0.38-inch long and 0.150-inch deep. Neither of these cracks were initiated or related directly to the ruptured piece. Low-powered microscopic examination exhibited a fine coating crack along the same length and arch of the small polycarbonate crack. The majority of the coating cracks appeared on the inner polycarbonate surface.



View Looking Forward. Note Transverse Coating Cracks.

Figure 43. Transparency Number C3/0043 After Bird Impact.



Forward

Coating Cracks in Impact Area

Figure 44. Transparency Number C3/0043 After Bird Impact.

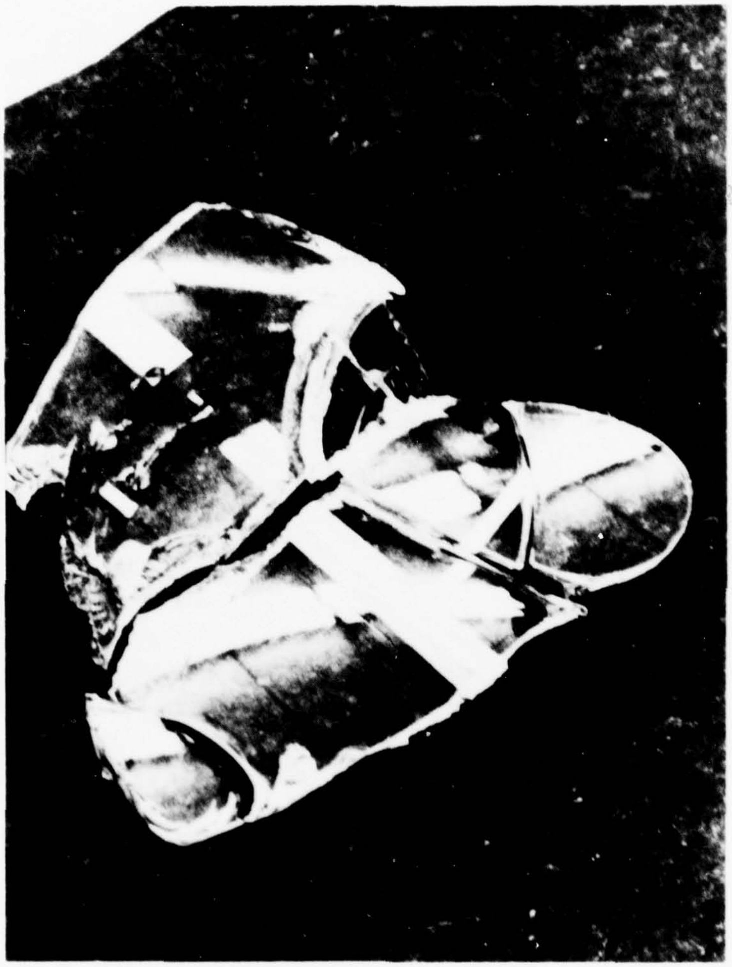


Figure 45. Polycarbonate Pieces from Transparency C3/0043.

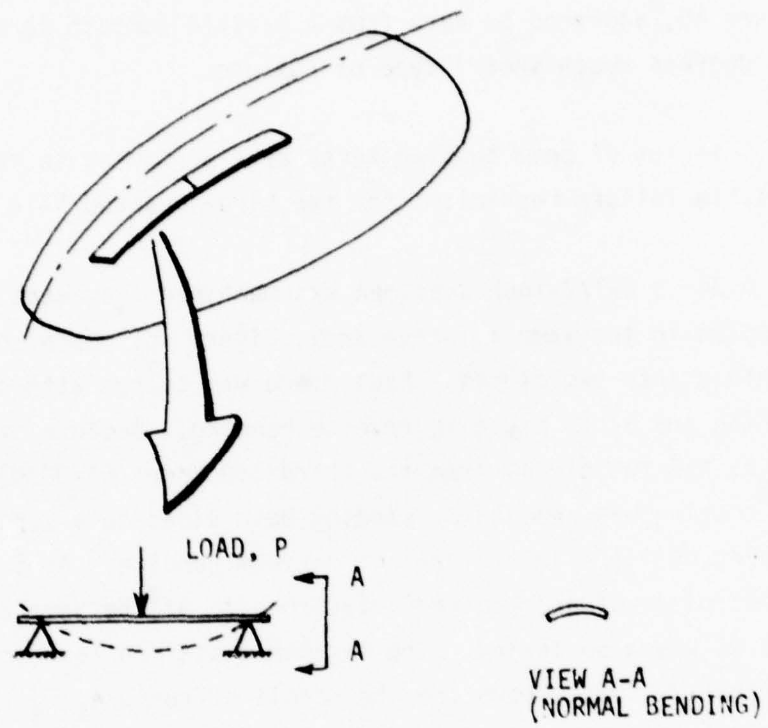
The ruptured transparency surface had indications of both ductile and brittle fracture. The suspected origin of the transparency failure, Figure 45, appeared to vary from a brittle (smooth tensile) to ductile (45 degrees rough shear) type of failure.

A series of beam bending tests were performed to further investigate possible failure mechanisms for the bird-impacted F-16 canopies.

A 24- x 2-1/2-inch specimen was machined from each of the three tested canopies in the same relative area, Figure 46. Each specimen was then machined into two pieces. Each piece was tested with one piece in normal bending and other piece in reverse bending. Because of similar maximum loads, the two pieces from the third specimen (C1/0030) were run after the coating was removed by sanding both sides to a depth of approximately 0.002/0.003-inch in the area of maximum bending. As a base comparison, a flat piece of as-received polycarbonate of the same approximate cross section was also tested. The maximum sustained load achieved was recorded in Table 8. Also shown are the moduli of rupture, F_B (the maximum achievable "stress" determined from $M_{MAX}C/I$), and the percentage of thickness reduction in the area under the load P (referred to as necking).

Results of these tests showed that:

- The transparency is approximately 5 percent stronger in normal bending than in reverse bending.
- Permanent necking occurs in these specimens, as contrasted to transparency failures under bird impact where no discernable permanent necking occurs. The amount of necking is more severe in the bare specimen and least severe in the coated specimen.
- Examination of all the ruptured coated specimens revealed a series of coating cracks which were all essentially parallel and normal to the direction of bending strain. The number of these cracks increases geometrically with the bending moment. These coating cracks extend into the polycarbonate in the area



Load, P, applied at maximum crosshead rate of 20 in./min.

Figure 46. Transparency Beam Bending Test Orientation and Specimen Location.

TABLE 8. BEAM BENDING TEST RESULTS

TRANSPARENCY NUMBER	COATED	SANDED	TYPE OF BENDING	F_B	P_{MAX}	NECKING	DEFLECTION AT FAILURE (IN.)
C4/0029	X		NORMAL	15.3 KSI	558 LBS.	-7%	-
C4/0029	X		REVERSE	14.7 KSI	534 LBS.	-9%	3.13
C2/0040	X		NORMAL	16.2 KSI	591 LBS.	-11%	-
C2/0040	X		REVERSE	15.4 KSI	561 LBS.	-2%	2.98
C1/0030		X	NORMAL	16.3 KSI	595 LBS.	-10%	-
C1/0030		X	REVERSE	15.4 KSI	561 LBS.	-6%	7.22
UNCOATED			NORMAL	16.2 KSI	589 LBS.	-13%	-

The failing "stress" (modulus of rupture) (F_B), the maximum load, and deflection are all normalized on the basis of the section modulus (I/C) of the first specimen. Hence the values recorded can be regarded as comparable unit values.

of maximum bending creating fissures leading to the specimen failure. All the coated specimens exhibited oriented molecular rearrangement or shear planes below the tensile cracks in both the polycarbonate and the coating. The shear planes intersect at approximately 45 degrees. Microscopic examination also revealed a series of highly oriented microcracks and fissures in various stages of development in the x-, y- and z-directions.

Figure 47 illustrates a section removed from the ruptured beam test specimen prepared for electron beam microscopic examination.

- The sanded specimens reached approximately the same maximum load as the coated parts. However, they achieved a significantly greater deflection before total collapse. The failure is attributed to a series of small crack fissures that interconnect. A series of crack fissures of various lengths and depth were oriented perpendicular to the applied strain. Molecular realignment, or slip planes, are also present in these sanded specimens. It appears that not all of the adverse effects of the coating were removed by the sanding even though the coating itself was totally removed.
- The uncoated polycarbonate specimen did not rupture during the test. The test was terminated because of excessive specimen deflection. No cracks or fissures were found. Highly oriented molecular realignment or molecular cleavage was present about the neutral axis and extended into the tensile surface.

A series of eight photographs associated with the coating crack analysis are shown in Figures 48 through 55.

Figure 48 shows coating cracks as well as an isolated crack in polycarbonate from a section of Transparency C3/0043. Figure 49 shows a crack that has originated in the coating and propagated into the polycarbonate.

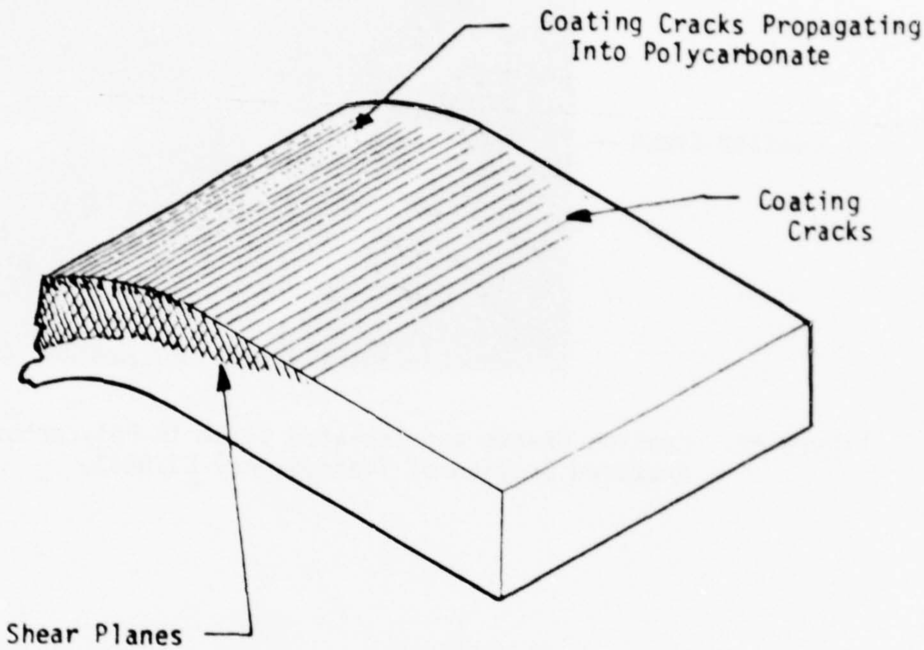


Figure 47. Section Removed From the Ruptured Beam Test Specimen Prepared for Electron Beam Microscopic Examination.

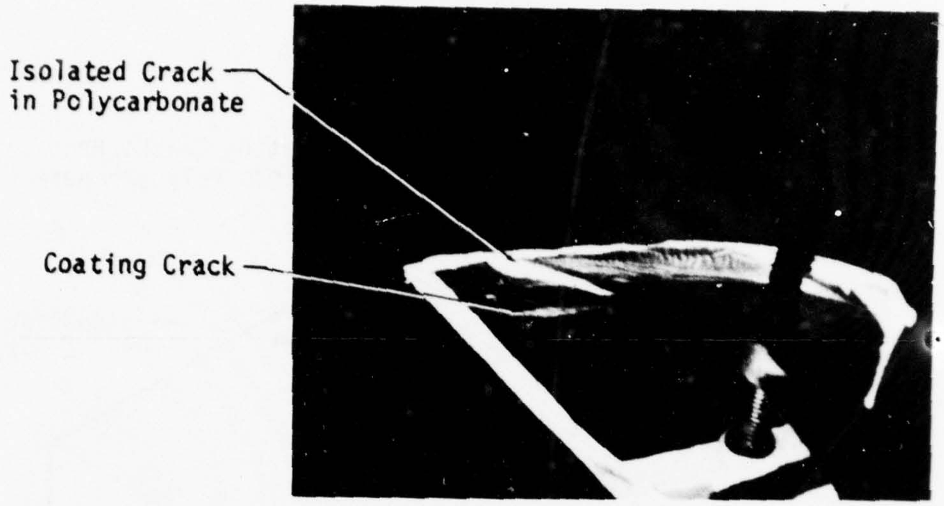


Figure 48. Coating Cracks and Isolated Crack in Polycarbonate From a Ruptured Section of Transparency C3/0043.

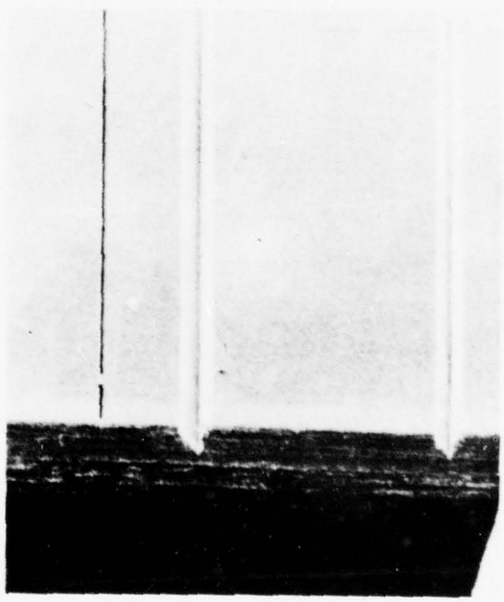


Figure 49. Top Side View, Depicting a Coating Crack and a Coating Crack Propagating into Polycarbonate Substrate in C4/0029 Transparency Beam Specimen (96 X).

Figure 50 shows delamination of the coating as well as an incipient crack fissure in a section of Transparency C3/0043. Figure 51 depicts a series of crack fissures in a specimen from the polycarbonate beam test. Figure 52, 53 and 54 display additional cracks in specimens from the beam tests. Figure 55 shows the coating thickness magnified 200 times.

Film hardness tests were conducted by the pencil test method in accordance with the test procedures specified in ASTM D3363-74. The pencil hardness tests were conducted on both coated and uncoated sections of C3/0043. The uncoated specimen was prepared by sanding 0.002-/0.003-inch material off the coated transparency section of C3/0043. The test results indicate that the surface of the coated transparency is harder than the surface of the uncoated transparency.

The test results are:

A. Coated transparency section = F Hardness

B. Uncoated transparency section = B Hardness

where the scale of hardness is

6B-5B-4B-3B-2B-B-HB-F-H-2H-3H-4H-5H-6H
SOFTER HARDER

As a further investigation of coated/uncoated polycarbonate hardness tests were conducted with the standard Shore "D" Durometer hardness testing apparatus on polycarbonate components and on as extruded MIL-P-83310 materials. The results shown below are on an average of six test areas:

PART/MATERIAL/I.D.		SHORE "D" HARDNESS
Transparency C3/0043	(Processed)	74
Transparency C3/0004	(Processed)	74
MIL-P-83310	(As Extruded)	72
Uncoated Specimen	(As Extruded)	72

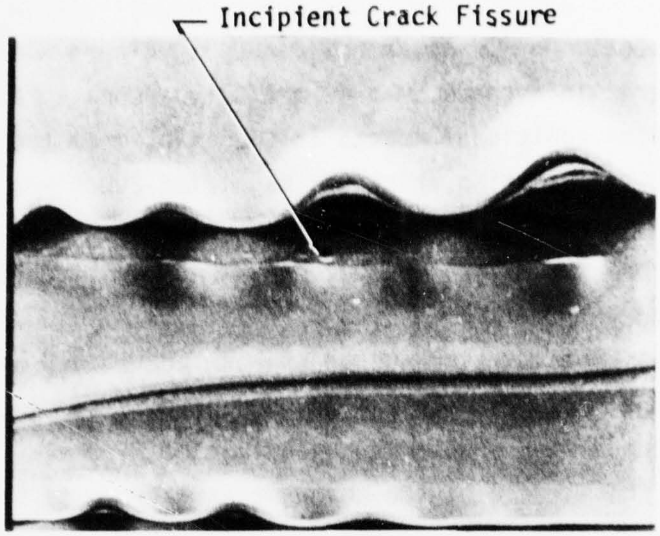


Figure 50. Coating Convolution and Delamination with Incipient Fissure Highly Oriented in the Plane of Coating Rupture (96 X).

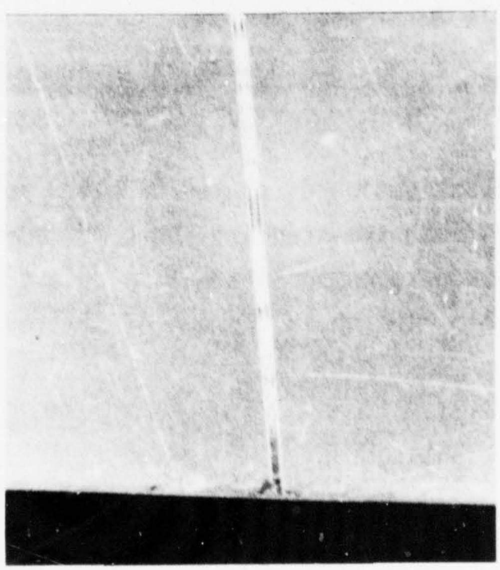


Figure 51. Top View of Coating Crack Depicting a Series of Highly Oriented Crack Fissures in Polycarbonate Beam Test Specimens (220 X).

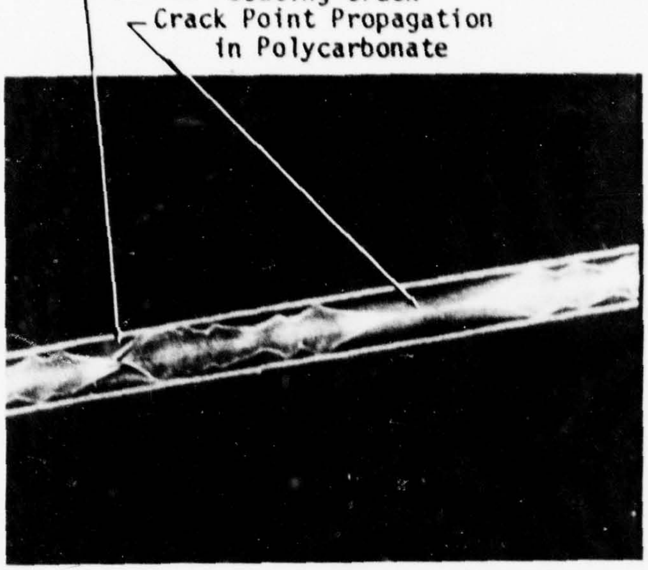


Figure 52. Close-up of Figure 51 Highly Magnified to Depict Crack Point Geometry and Crack Propagation in Polycarbonate Beam Test Specimen (1100 X).

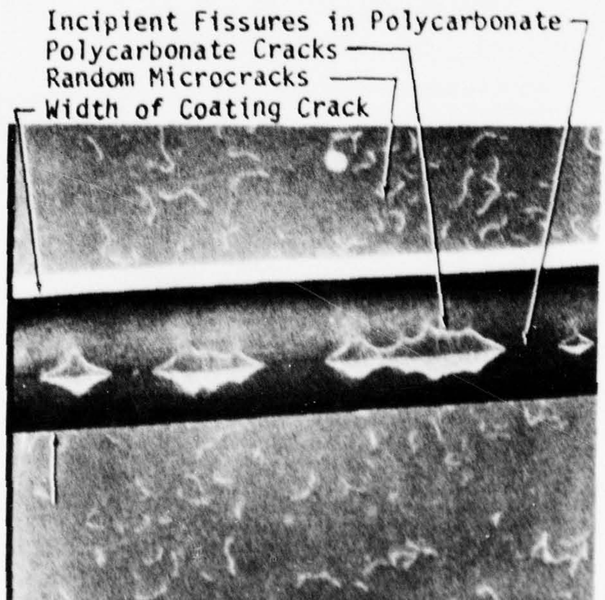


Figure 53. Random Unoriented Microcracks in Coating and Coating Microfissures in Polycarbonate Beam Test Specimens (440 X).

Width of Coating Crack

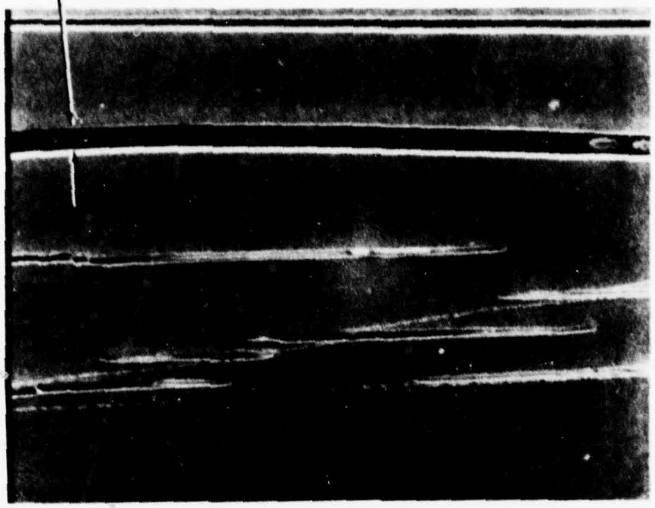


Figure 54. Assortment of Coating Cracks of Various Lengths and Widths with Microcracks in the Polycarbonate Beam Test Specimens (88 X).

Thickness of Coating
Magnified 200 Times.

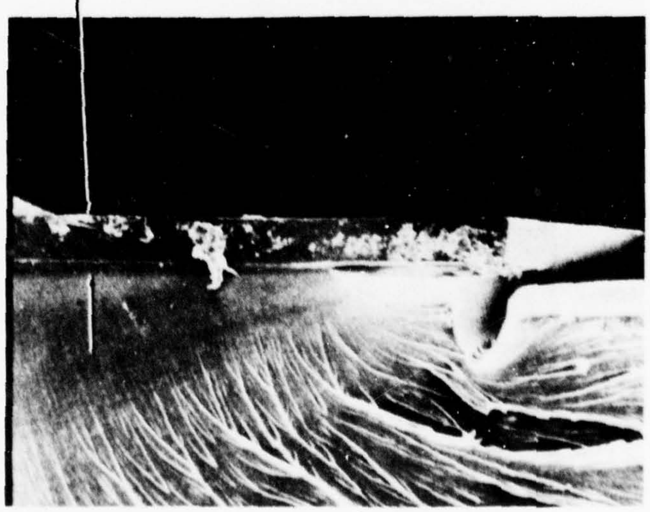


Figure 55. Protective Coating Thickness Determination Performed on a Freshly Broken Specimen from Transparency C3/0043 (200 X).

DESIGN APPLICATIONS

A goal of the Windshield Technology Demonstrator Program (WTDP) was to define potential design changes to the current F-16 concept that would meet the various impact test levels of 350, 400, 430, 450, 500, and 562 knots. Preliminary thicknesses were to be based on the bird impact test results and a bird impact math model and computer program would be utilized to finalize the required thicknesses after appropriate correlation was accomplished with the test data.

The results of the fifteen tests conducted at ambient temperature on 0.50-inch and 0.62-inch thick coated transparencies, and on a 0.75-inch transparency tested by General Dynamics, were plotted to produce the pass/fail curve shown in Figure 56. The curve assumes a simple progression and predicts the polycarbonate nominal thickness required for a coated F-16A transparency to defeat a four-pound bird at a selected velocity (ambient temperature, impact Point A). The curve indicates that a 0.76-inch transparency would defeat a four-pound bird at 350 knots, and a 1.00 inch transparency would defeat a four-pound bird at 562 knots. This curve is based on coated polycarbonate, which appears to have a much smaller tensile elongation capability than uncoated polycarbonate. The slope of the curve indicates that the change in velocity varies with the thickness change raised to the 1.73 power.

Tables 9 and 10 show the deflection of the transparency at the centerline due to bird impact, which was recorded by cameras and plotted by AEDC/ARO. These values are not considered to be accurate below a deflection of 0.50-inch because the transparency thickness inhibits the deflection measurements.

Figure 57 is a plot of deflection at the eye point (Sta. 140) for 0.50-inch transparencies and for 0.62-inch transparencies due to four-pound birds. These curves were used to develop a deflection versus thickness curve for four-pound birds at 350 knots, Figure 58. Extrapolation of this curve indicates that a 0.68-inch transparency would be required

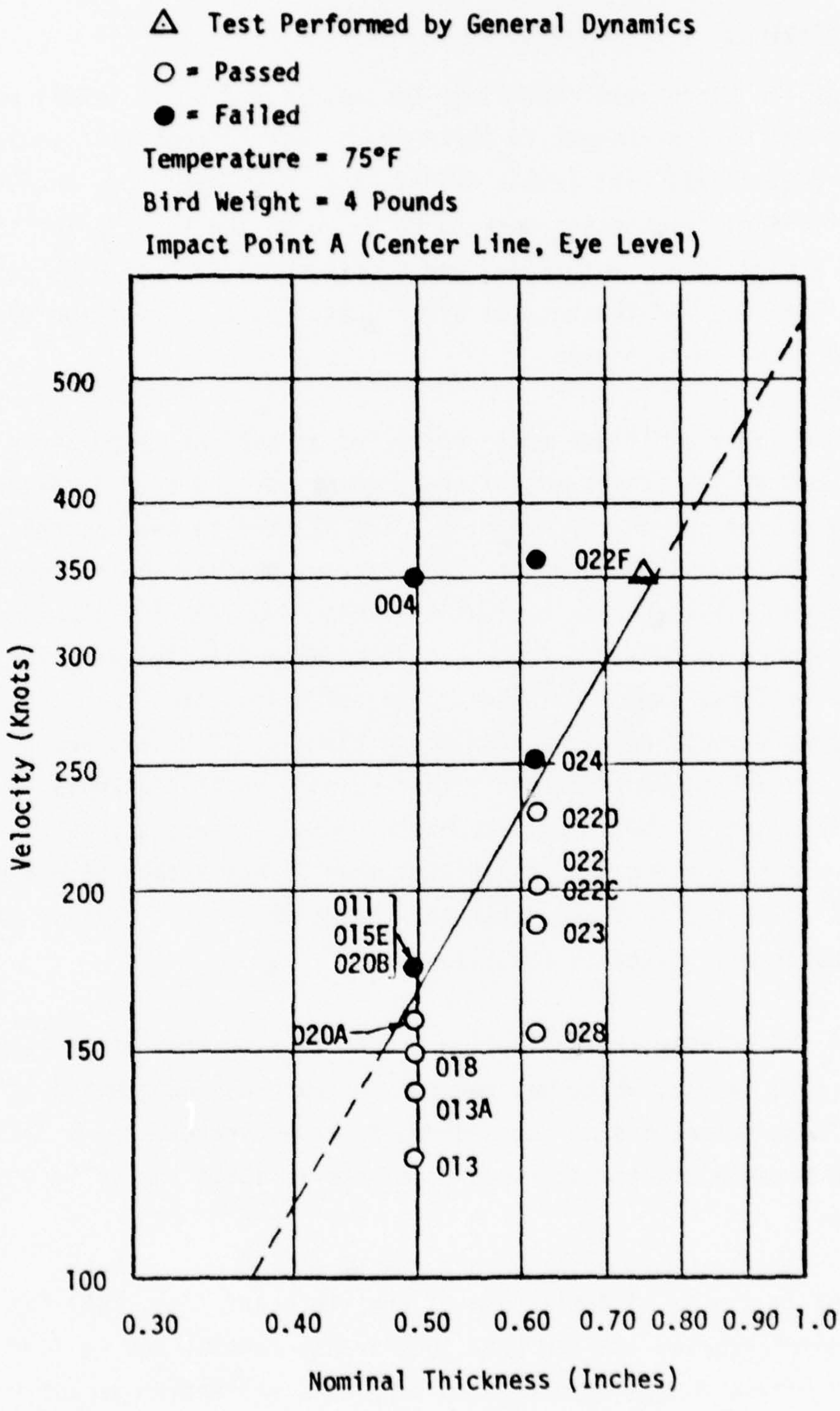


Figure 56. Pass/Fail Curve for Bird Impact (Coated Polycarbonate Transparencies).

TABLE 9. DEFLECTION FOR A 0.50-INCH TRANSPARENCY

Test No.	V (Knots)	Remarks	Deflection (Inches) Maximum	Deflection (Inches) Sta. 140 (Eye Point)
001		Static - 2200 Lb.	0.991	
002		Static - Hot - 2200 Lb.	0.745	
003		Static - Cold - 2200 Lb.	0.778	
013	124		0.8	0.30
013A	140		1.2	0.38
018	149		1.2	0.54
020A	158		0.9	0.46
GD4	158	Uncoated	2.3	0.30
014A	159	Hot	1.2	0.62
015A	165	Cold	0.8	0.70
020B	173	Failed Aft	1.7	0.77
015B	173	Hot	2.9	0.38
015E	174	Failed Aft	2.3	1.0
011	176	Failed Aft	1.2	1.0
012	191	2 Lb. Bird	1.0	0.3
GD3	340	Uncoated	8.3	6.4
006	344	2 Lb. Bird - Failed Aft	4.6	2.9
GD1	344	Uncoated - 2 Lb. Bird	5.4	1.7
004	349	Failed Forward	1.6	---
GD2	351	Uncoated - 3 Lb. Bird	6.9	3.9
GD5	362	Uncoated	8.9	7.7

NOTES: (1) Temperature: 75°F

(2) Impact Location A

(3) Bird Weight: 4 Lbs. unless noted otherwise

TABLE 10. DEFLECTION FOR A 0.62-INCH TRANSPARENCY

Test No.	V (Knots)	Remarks	Deflection (Inches)	
			Maximum	Sta. 140 (Eye Point)
021		Static - 2200 Lb.	0.391	
028	155		0.8	0.31
023	188		1.9	0.77
022B	199	Cold - Transparency Frosted		
022A	200	Hot	2.5	
022C	201	HUD Installed	1.2	0.38
022	202		2.2	0.62
022D	230		2.3	0.50
022E	231	"B" Location	2.6	0.62
028A	233	"B" Location-Failed Aft	2.3	0.50
022F	363	Failed Aft	5.4	2.46

NOTES: (1) Temperature: 75°F

(2) Impact Location A

(3) Bird Weight: 4 Lbs.

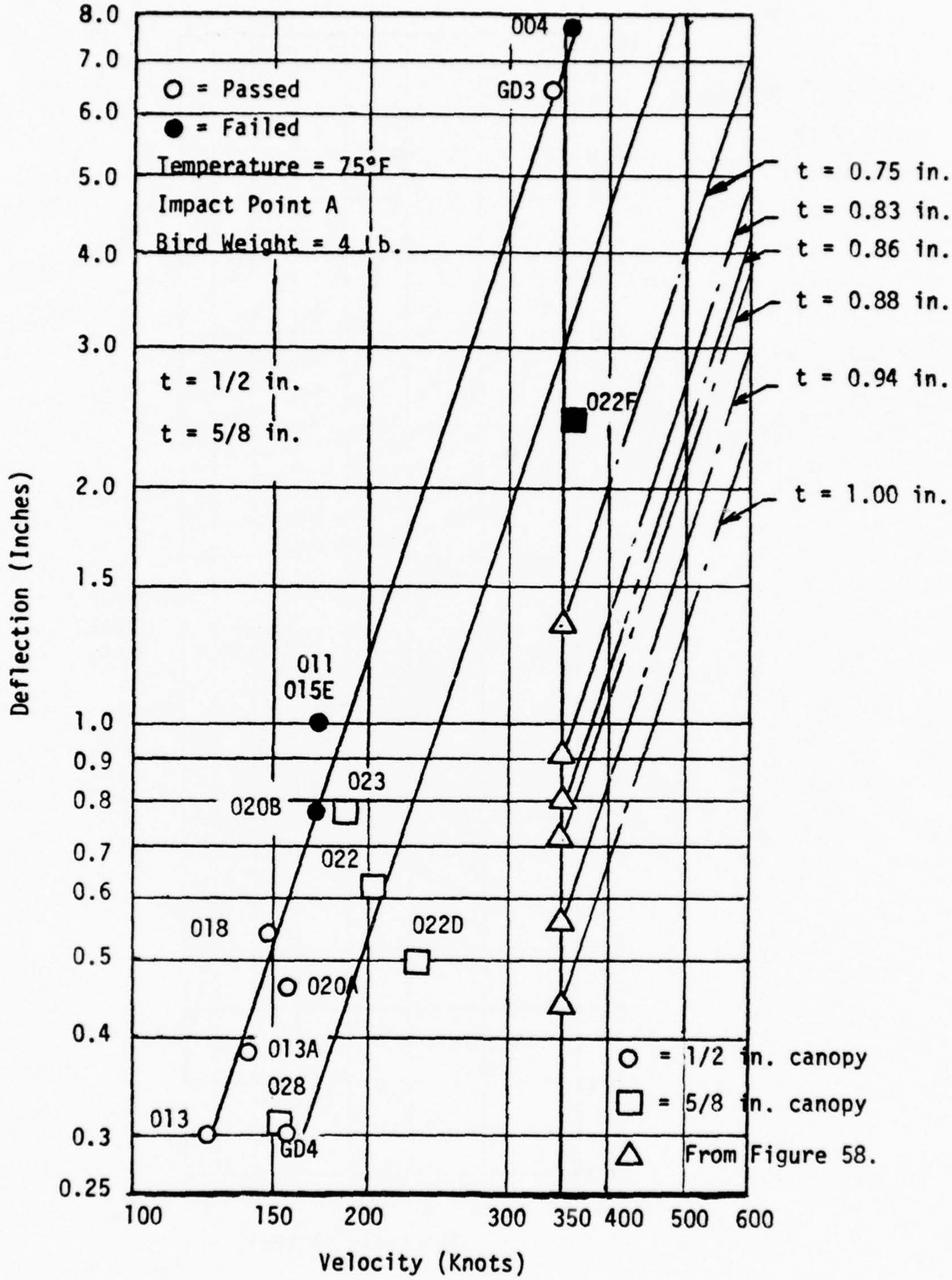


Figure 57. Deflection at Eye Point Due to Bird Impact.

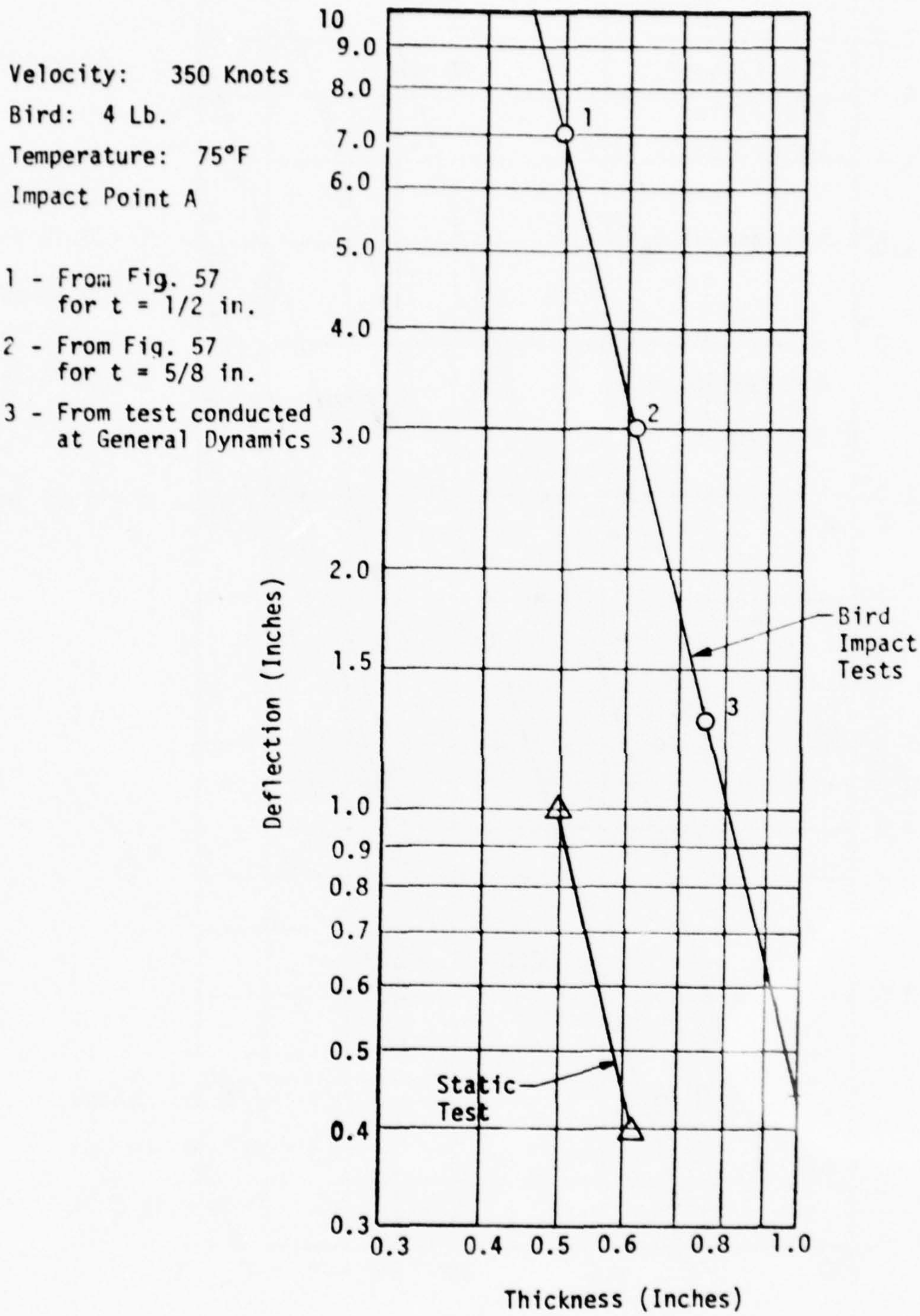


Figure 58. Deflection at Eye Point Due to Bird Impact.

to limit the eye point deflection to less than 2.00 inches. The static test deflection at Sta. 112.5 is also shown on this figure. The curves (dynamic and static) are near parallel and indicate that the deflection/thickness ratio is similar for both cases. The slope of the curve in Figure 57 indicates that change in deflection varies with the change in velocity to the third power. The slope of the curve in Figure 58 indicates that the change in eye point deflection varies with the inverse of the thickness change to the fourth power.

Table 11 is a compilation of the transparency nominal thicknesses and expected deflections, for coated monolithic transparencies, required to sustain the impact of a four-pound bird at Location A for six selected velocities. The thicknesses were extrapolated from the pass/fail curve of Figure 56. The expected deflection at eye point for the individual cases has been determined from Figure 57. These deflections were extrapolated from the deflections at 350 knots as determined in Figure 58. It is assumed that the load deflection ratio is identical for all transparency thicknesses.

Table 11 also lists a modified thickness necessary to sustain the impact of a four-pound bird at Location B. The pass/fail curve, Figure 56, indicates that a penetration velocity for a 0.62-inch transparency is 245 knots. Test results (Test 022E and 028A) indicate that the penetration velocity for impact at Location B is 232 knots. Therefore, a factor of 3 percent has been applied to the thicknesses defined for impacts at Location A, since Location B, based on limited test results, appears to be less capable of sustaining bird impacts than Location A. Location B is approximately 14 inches closer to the forward edge of the canopy.

BIRD IMPACT TESTING CONCLUSIONS

The primary goal of this test effort was to provide data that were applicable to the design of a bird proof polycarbonate canopy for the F-16 aircraft at velocities up to Mach 0.85. The first canopy tested was a

TABLE 11. THICKNESS REQUIREMENTS AND DEFLECTIONS FOR DESIGN VELOCITIES

VELOCITY (1) (KNOTS)	THICKNESS REQUIRED (INCHES) (2)		DEFLECTION (3) AT PILOT'S HEAD (INCHES) (FROM FIGURE 57)
	IMPACT AT LOCATION A (FROM FIGURE 56)	IMPACT AT LOCATION B	
350	0.76	0.78	1.30
400	0.83	0.85	1.35
430	0.86	0.89	1.45
450	0.88	0.91	1.55
500	0.94	0.97	1.65
562	1.00	1.03	1.80

(1) Bird weight 4 pounds
Temperature 75°F

(2) Tolerance = $\pm$ 0.03 in.

(3) Deflection due to impact at Location A.
Tolerance = $\pm$ 0.15 in.

production, 0.50 inch thick, coated, polycarbonate canopy. The canopy unexpectedly failed on the first test. This identical canopy configuration, except uncoated, was tested by General Dynamics in the early bird impact development tests, and did not fail. Because of the failure of the first canopy the subsequent series of tests for this program had to be directed toward making a determination of the effect the protective coating had on the canopy bird impact resistance. It was determined that the potential for canopy failure as the result of bird impact is greater on a coated polycarbonate canopy because it has a lower tension elongation capability than an uncoated canopy. The analysis previously presented in this section shows that the failure originates in the hard coating and propagates into the polycarbonate. Additional studies showed that an uncoated canopy deflects more than twice as much as a coated canopy when tested under identical conditions. The strain measurements for these tests were similar and the deflections were substantiated with static tests on small beam segments. The increased stiffness of the coated canopy was attributed to thermal embrittlement, as the result of applying the coating, or the addition of the hard coating to the surface of the softer polycarbonate material.

A graph (Figure 56) was derived from this series of tests to provide the thickness of coated polycarbonate required in an F-16 canopy to defeat a four-pound bird at velocities up to Mach 0.85 (560 knots) when impacted on the canopy centerline at a point in line with the pilot's design eye position. A deflection curve (Figure 57) indicates that these thicknesses will limit deflection of the canopy to less than 2.0 inches over the pilot's head. These data are used in Section XI to design a laminated canopy concept for application to the F-16 aircraft.

The tests conducted to investigate the effect of temperature on the impact capability of a monolithic polycarbonate canopy indicated that the hot or cold canopy has a greater resistance to bird impact than the canopy tested at 75°F. These results are substantiated by high-strain-rate tests on polycarbonate material (Reference 18), which show that polycarbonate material tested at 75°F has a lower relative toughness than at hot or cold temperatures.

The effect of temperature on canopy deflection was not consistent. Under static load, the canopy tested at 75°F deflected about 30 percent more than when tested at hot and cold temperatures. Under bird impact loads, the deflections were higher for the heated canopies. The 5/8 inch thick canopy deflected 14 percent more when hot, and the 1/2 inch canopy varied: one test showed equal deflections and the second canopy deflected 30 percent more when hot than at 75°F. Measuring canopy deflection as the result of bird impact at cold temperatures was impeded by frosting of the canopy. The static deflections were recorded by using deflectometers.

There is a need for improvements in the methods of measuring deflection due to impact loads at cold temperatures and to improve the accuracy of measurements at low amplitudes.

It was established that tension strain under bird impact and static loads was slightly higher for high temperatures but significantly lower for cold temperatures.

The results of two similar bird impact tests on the same canopy were compared to determine the effect of multiple impacts. The results were similar since the strains were in the elastic range for polycarbonate material (Reference 18). This indicated that the strains in the elastic range, due to bird impact loads, do not increase with identical subsequent impacts; consequently, there appears to be no significant degradation in impact capability of the material.

Tests were conducted to define sensitivity to impact location. The limited tests performed indicated that a centerline birdstrike is more critical to canopy survival than an off-center impact, and that the energy absorption capability of a high on-center impact is approximately 10 percent greater than a low impact. Deflections in the area of the pilot's head were not significantly different for the high and low impacts.

It was established that contact of the canopy with the HUD assembly does diminish the overall canopy deflection.

The effort to establish the maximum velocity for bird impact capability of the F-16 canopy using 2 and 3 pound birds was not accomplished, since the major thrust of the test program, after the initial failure, was directed toward defining the capability of the existing F-16 canopy to resist a four-pound bird.

Graphs showing canopy deflection and strain versus time were prepared from the test data for the purpose of correlating the bird impact computer program.

Canopy thickness measurements revealed material thinning. Maximum thinning occurred along the centerline in the area of the pilot's head and must be considered when designing a canopy of this shape. Successful testing and application of test results is affected by repeatability of test conditions and consequently the control of material thickness is necessary.

Variation in bird density, trajectory, attitude of bird at impact, and impact point affect test repeatability and results. The use of a chicken as the test projectile causes delays in testing due to the necessity of cleaning the test area between shots. The accuracy in test time would be improved if a suitable substitute could be found to replace the chicken as the test projectile.

SECTION VIII
TRANSPARENCY EDGE JOINT TEST

This section presents a discussion of edge joint tests and test results that were conducted by Douglas to verify the adequacy of the attachment system for a laminated transparency concept adaptable to an F-16 aircraft. The basic configuration selected for the F-16 alternate canopy design was a three-ply laminated transparency comprised of an acrylic outer ply and a polycarbonate structural ply joined by an interlayer. The spacing between bolt holes, edge distance, and hole bushings are identical to those of similar edge tests conducted by Texstar for the F-16 program (Reference 22). The loads are consistent with the loads documented in Reference 22, but the temperatures and number of pressure cycles were revised. The test setup was designed to simultaneously test 18 specimens under the calculated loads and temperatures. Laminated configurations from two vendor sources were tested.

Three types of tests were conducted to verify the structural integrity of the proposed laminated transparency: cyclic load tests at low, ambient and elevated temperatures, cyclic thermal tests at constant load, and a high temperature test at twice ultimate load.

A series of 4.0 x 7.0 inch flat test specimens were assembled in two parallel chains, enclosed in an environmental chamber to permit application of high and low temperatures, and loaded simultaneously by a hydraulic jack.

The goals of this test were to evaluate the edge attachment area of laminated transparency designs for materials from several sources and to verify that the materials and the edge attachment area of the laminated transparency designs would sustain without structural failure or delamination the equivalent of four lifetimes (8000 hours) of internal pressure cycling at aircraft operating temperatures based on a 2000 hour transparency design life. These goals were achieved.

BACKGROUND

The prime purpose of an edge attachment is to join an aircraft transparent enclosure to the primary structure and to transmit the developed loads from the transparent material to the airframe. The edge attachment area of a transparent enclosure is exposed to a harsh environment and must be capable of withstanding extreme temperature gradients, cyclic loads, and stress concentrations. An inadequate edge attachment design will result in a reduced service life and high life cycle costs.

Two failure modes prevalent in windshields and canopies are delamination and ruptured attachment holes. Delamination has been defined as a separation between two plies, usually between the outer face ply and the interlayer material.

One cause for delamination is attributed to the vast difference in the expansion/contraction rates between the structural plies and interlayer materials when subjected to temperature extremes. Edge restraints and cyclic pressurization loads can also cause delaminations. Potential methods for minimizing delamination is to use a retainer at the edge of a laminate and/or select one of the more recently developed high elongation silicones as an interlayer material.

Ruptured attachment holes are identified as an elongation along one edge of the hole or the material is torn out at the transparency edge. Ruptured attachment holes have been caused by excessive bearing loads, thermal stresses, or a reduction of the material strength as a result of crazing.

Crazing has been defined in Reference 23 as fine cracks which may extend in a network on or under the material surface or through plastic materials. Improperly drilled attachment holes will produce stress crazing in polycarbonate material at low stress levels. When subjected to elevated

temperatures, polycarbonate is exceptionally prone to crazing. Bolt Hole drilling methods and hole finish must be controlled because crazing of polycarbonate can significantly lower the impact strength of the material. Reference 24 notes the following requirements for proper hole drilling: keep the part cool by using slow to medium cutting speeds and feed rates, use a coolant, and maintain a sharp cutting edge at all times.

Other factors to be considered are hole spacing and hole diameter. A minimum number of bolts is desirable to enhance maintainability; however, bearing stress is inversely proportional to the number of bolts and must not exceed the critical stress level for polycarbonate.

Edge designs greatly affect the costs associated with interchangeability, including initial tooling and maintenance costs. Therefore, consideration must be given to bolt hole size (loose versus tight holes), hole sealing (wet or dry), the type of bushing used to distribute the load, and adequate edge distance.

The initial F-16 production canopy was 1/2-inch coated monolithic polycarbonate transparency. The attachment holes were bushed with polycarbonate bushings. The edge attachment design for the initial canopy was successfully tested and documented in Reference 22. The test conditions included a tensile test which loaded each specimen to design load, ultimate load or to failure whichever occurred first. A series of specimens were subjected to tensile cyclic loading at high, low and ambient temperatures for a total of 1000 cycles.

The tests described in subsequent paragraphs relate to laminated specimens and monolithic specimens taken from canopies that had been subjected to other prior tests.

TEST SPECIMEN DESCRIPTION

Table 12 is a pre-test description of the test specimens which are shown in Figures 59, 60 and 61.

The uncoated laminated specimens were comprised of an 0.08 inch as-cast acrylic outer ply and a polycarbonate structural inner ply separated by an interlayer material. The polycarbonate selection was 0.62, 0.75 and 0.87 inch thick. Two types of interlayers were used: 0.10 inch thick (CIP) silicone, and a 0.03 inch thick copolymer. The copolymer is a Sierracin product designated as S-130. Two monolithic coated specimens, Figure 61, were taken from a canopy that had previously been bird impact tested for the Windshield Technology Demonstrator Program.

The laminated specimens were manufactured by two vendors. Sierracin fabricated six laminated specimens using the S-130 copolymer interlayer (-537, -539, -541, -543, -545 and -547), and six of the laminated specimens that used S-100 silicone interlayer (-521, -523, -525, -527, -529, and -531). Swedlow fabricated four laminated specimens with silicone interlayer (-523, -525, -529, and -531).

The glass filled polycarbonate bushings defined in Figures 62 and 63 were based on bushings defined in Reference 22. These bushings were installed in oversize holes (0.019 inch maximum to 0.010 inch minimum) and bonded in place with RTV630 after being primed with SS-4120. The oversize hole in the bushing provided bolt clearance of 0.025 inch minimum to 0.030 inch maximum.

The attachment holes were drilled through the face ply and structural ply at one end of the specimens. At the other end the face ply was cut back. The bolt holes were drilled by the vendors using their proprietary methods that are intended to prevent crazing in the bolt holes.

The fasteners were installed dry and were torqued to 25-35 in.-lb. for the 3/16 inch diameter screws and 69-80 in.-lb. for the 1/4 inch diameter bolts. No sealant was used in the holes.

TABLE 12. PRE-TEST SPECIMEN DESCRIPTION

CONFIGURATION OF Z5943266	MFG (1)	TYPE OF CONSTRUCTION	TYPE OF INTERLAYER	HOLE SIZE (IN.)	THICKNESS OF POLYCARBONATE	TAPE UNDER BOLTS (4)	REMARKS
-521	SK	Laminated	Silicone	1/4	0.62		(6)
-523	SK		(0.100 In.)	1/4	0.72		(5) (6)
-523	SWU			1/4	0.70	Yes	(6)
-525	SK			1/4	0.94	Yes	(6)
-525	SWU			1/4	0.87		(6)
-527	SK			3/16	0.62		
-529	SK			3/16	0.75		
-529	SWU			3/16	0.70	Yes	
-531	SK			3/16	0.94		(5)
-531	SWU			3/16	0.87	Yes	
-533	TEX	Monolithic (2)	-	1/4	0.50		
-535	TEX	Monolithic (2)	-	3/16	0.50		
-537	SK	Laminated	S-130 (3)	1/4	0.62		(6)
-539	SK		(0.030 In.)	1/4	0.75		(6)
-541	SK			1/4	0.87		(6)
-543	SK			3/16	0.62		
-545	SK			3/16	0.75		
-547	SK			3/16	0.87		

(1) SK = Sierracin, SWU = Swedlow, TEX = Texstar
(2) Specimens taken from canopy impacted as part of bird test program
(3) S-130 - Sierracin Copolymer
(4) 3M Polyurethane tape used to reduce crazing. Installed between polycarbonate and aluminum plates.
(5) Cracks in acrylic ply emanating from holes drilled in end without retainer.
(6) Washers were installed under head of bolts (acrylic side) to eliminate bushing pull out.

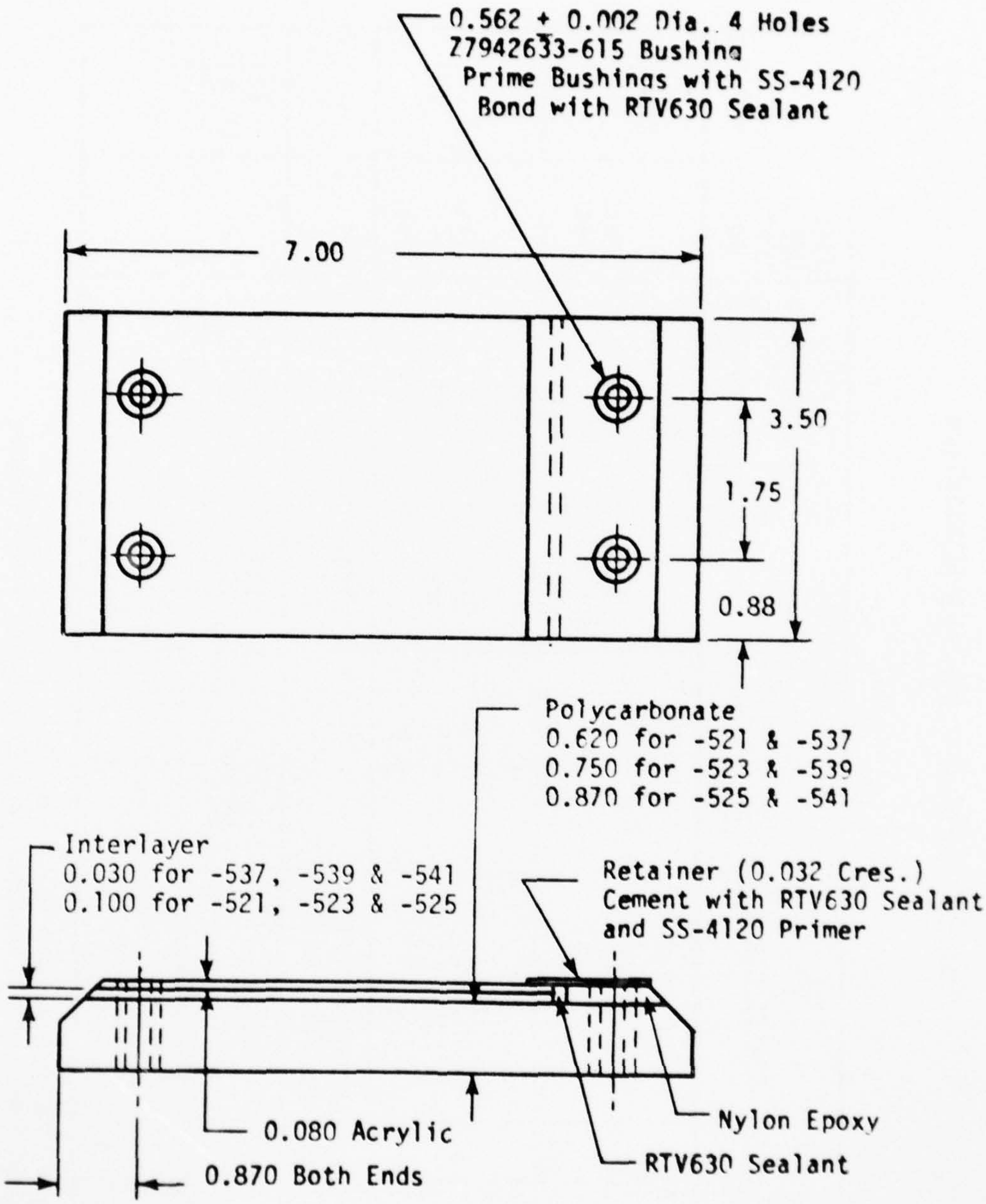


Figure 59. Specimens for 1/4 Inch Bolts

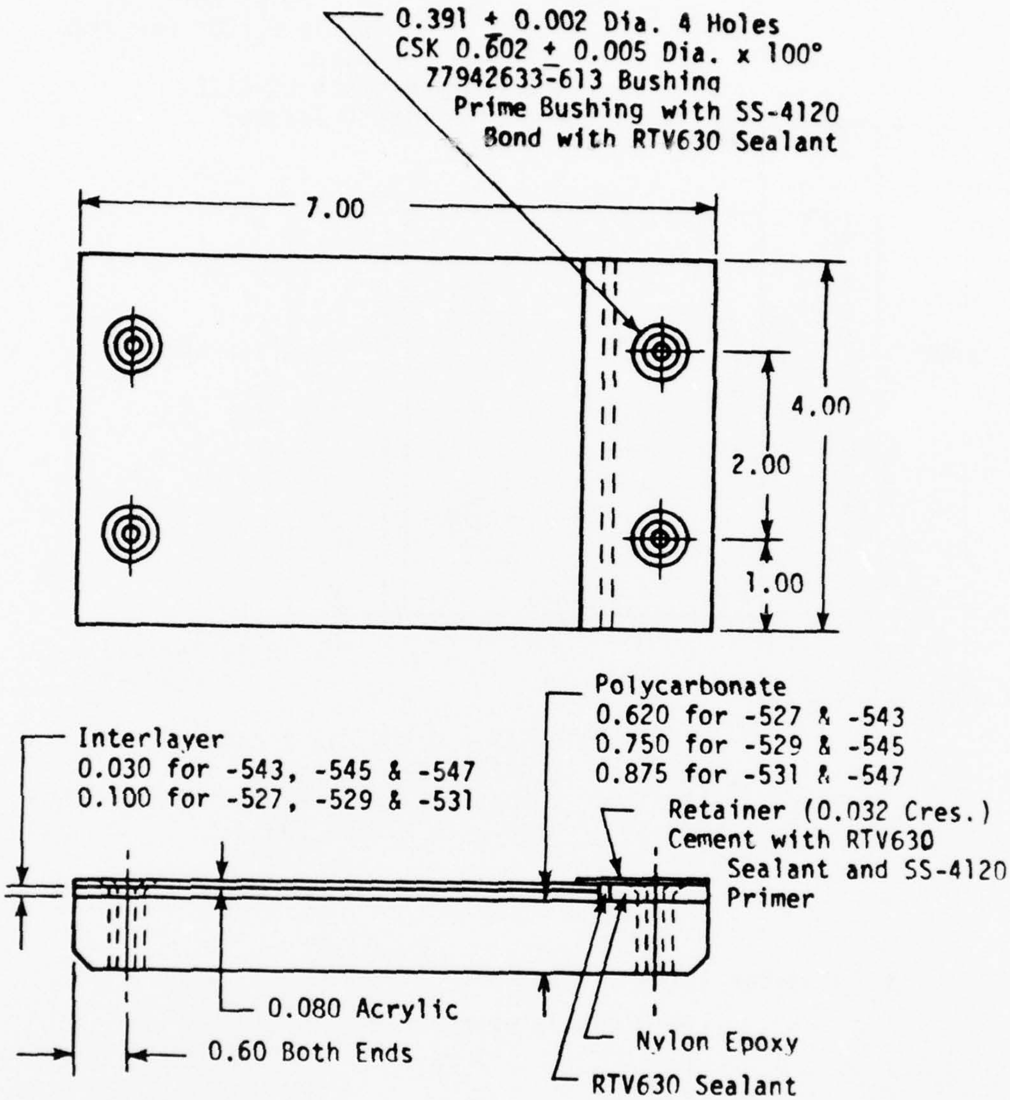


Figure 60. Specimens for 3/16 Inch Bolts

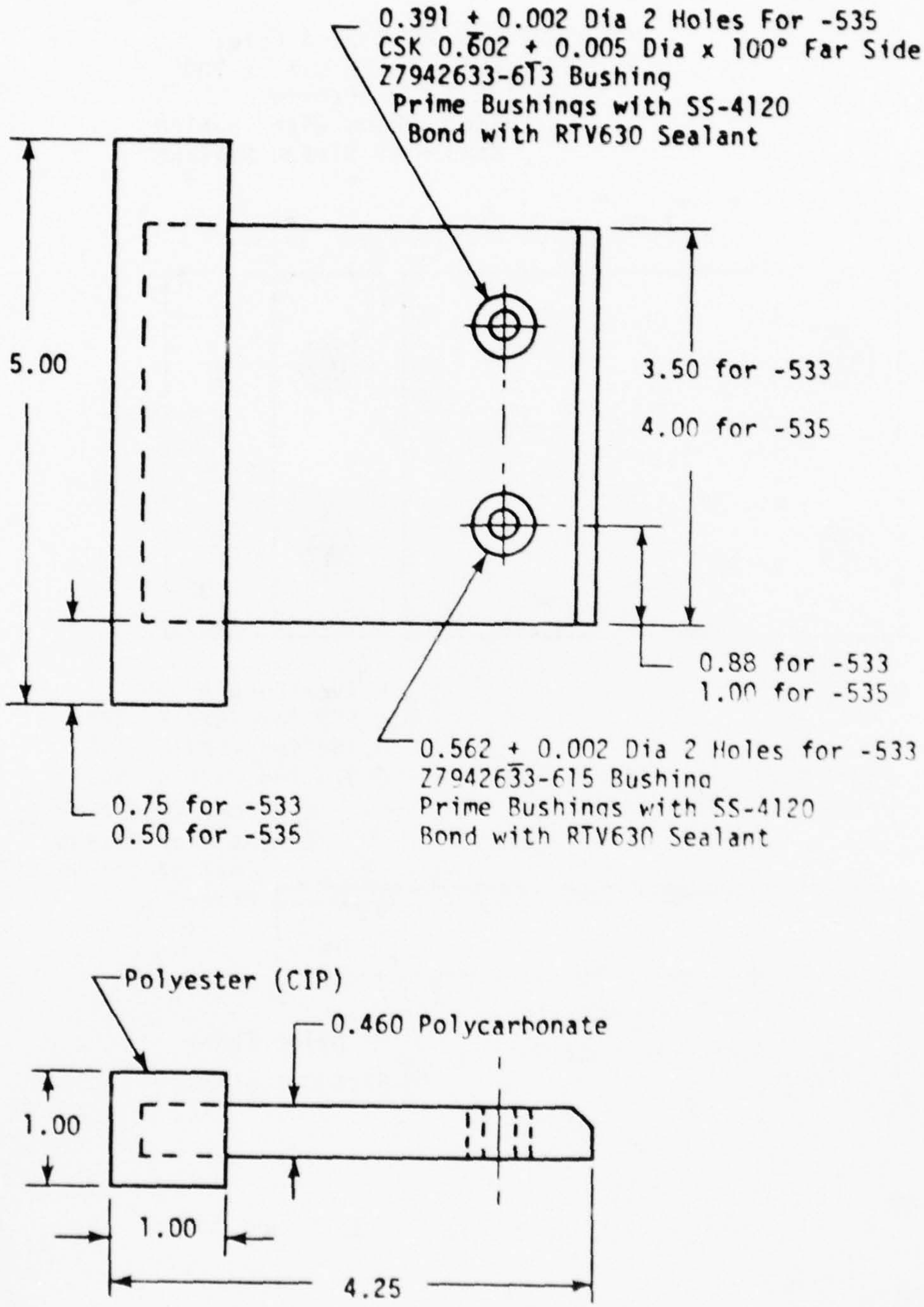
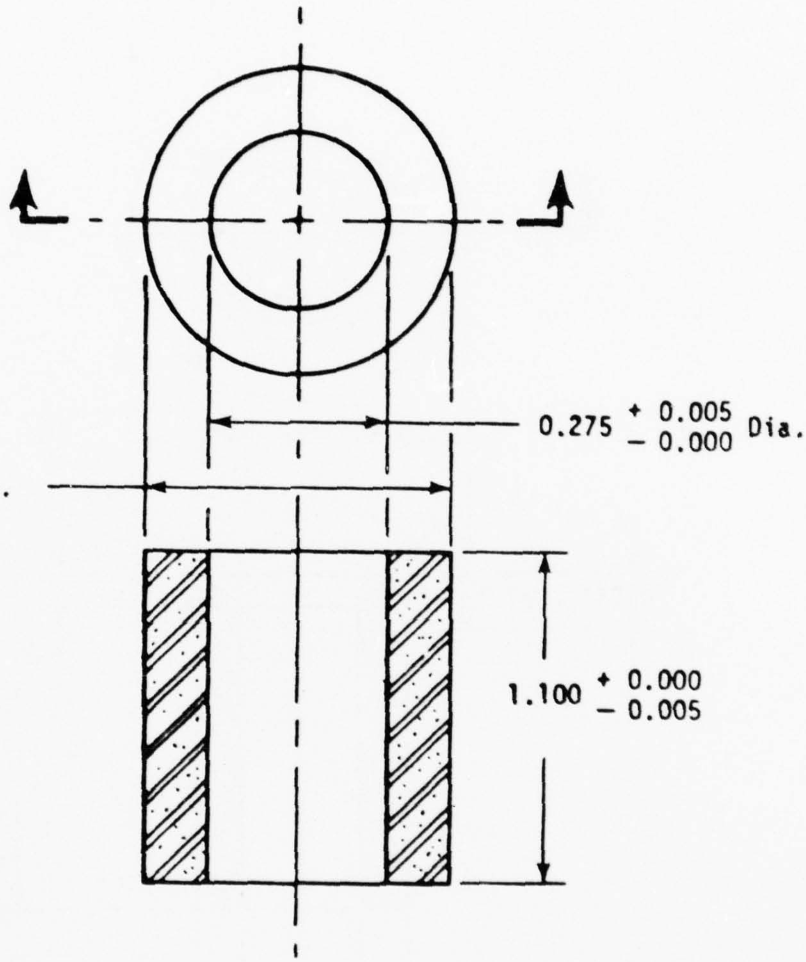
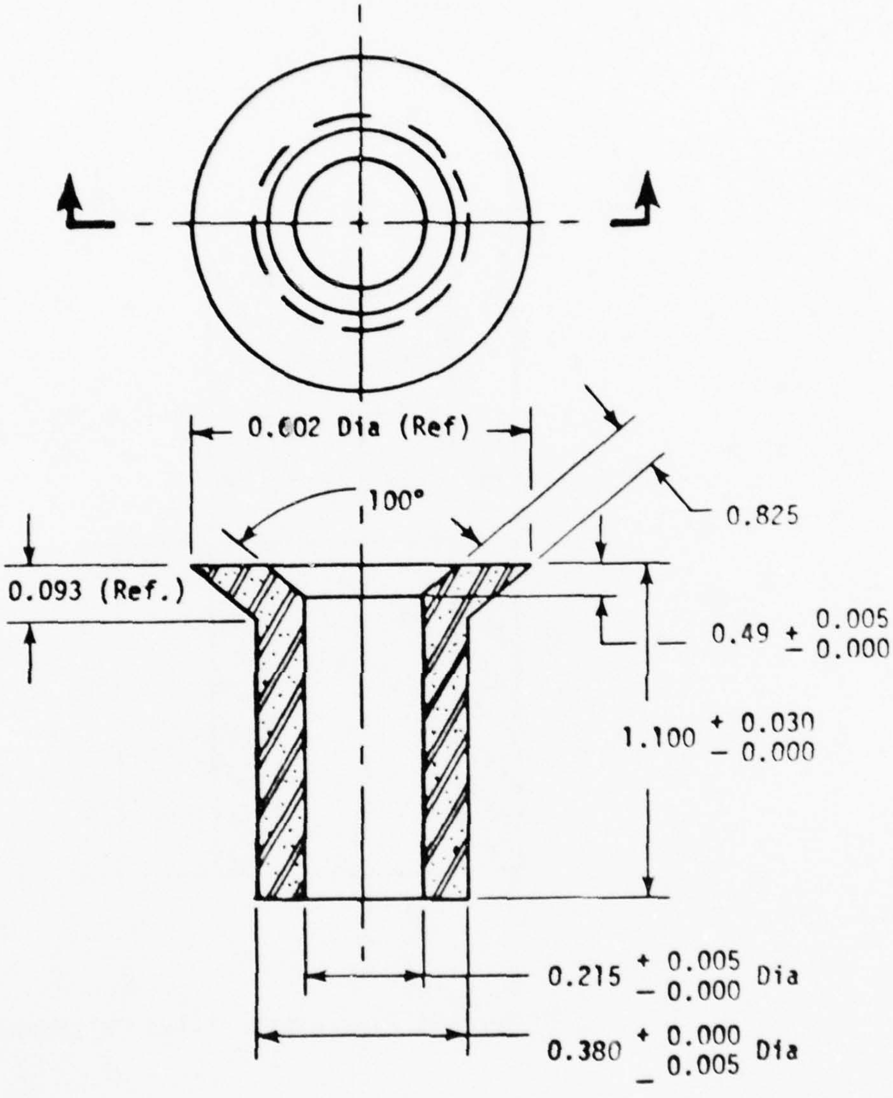


Figure 61. Monolithic Polycarbonate Specimens



20 Percent Fiberglass Filled Polycarbonate.

Figure 62. Bushing for 1/4-Inch Diameter Bolt. (Z7942633-615)



20 Percent Fiberglass Filled Polycarbonate.

Figure 63. Bushing for 3/16-Inch Diameter Bolt. (Z7942633-613)

TEST DESCRIPTION

Three types of tests were conducted on sixteen laminated acrylic/interlayer/polycarbonate specimens and two monolithic polycarbonate specimens as follows:

- Cyclic load/constant temperature, Test 1, 2 and 3, Table 13.
- Cyclic temperature/constant load, Test 4 and 5, Table 14.
- Cyclic temperature/twice ultimate load, Test 6, Table 14.

Loads equivalent to a 2P ultimate condition were applied and were based on a lifetime of 1814 cycles for four lifetimes (7256 cycles). Temperatures ranging from -65°F to 245°F were imposed in the specimens. The load was doubled for the final test. These loads, cycles and temperatures were based on operating conditions for an F-16 aircraft (Reference Section V).

Test Setup

The eighteen test specimens were assembled into two strings of nine specimens each and were bolted together as shown in Figures 64 and 65. The two strings were connected by a loading bar at each end and installed in the test fixture. The loading bar was designed to divide the input load (1191 lbs) into two loads (735 lb. and 456 lb.) as shown in Figure 66.

A pneumatic loading jack was attached to the test specimens, Figure 67, supported in an erector set type test frame made from 12-inch wide-flange beams. A clock timer triggered a solenoid valve in a pneumatic loading system which introduced pressurized air to the loading jack for a given time period. The load level was established by a pressure reducing valve in the air input line to the loading jack. The time rate of loading and unloading was established by a needle valve throttle adjustment of a special solenoid valve in each in/out air line. An electronic load cell installed in the load link with a strip-chart recorder read-out provided the load versus time trace of the cyclic load test. A mechanical counter in each test link recorded the number of load cycles. A polysterene environmental

TABLE 13. CYCLIC LOAD TEST (ONE LIFETIME)

TEST NUMBER	SPECIMEN CONSTANT TEMP. (°F ± 10°F	CYCLIC LOAD (Lbs.)(1)	CYCLIC LOAD (Lbs.)(2)	NUMBER LOAD CYCLES	TIME LOAD ON (Min.)	TIME LOAD OFF (Min.)	TOTAL TEST TIME (Hours)
1	-5	735	456	916	4.5	1.5	91.6
2	75	735	456	874	4.5	1.5	87.4
3	160	735	456	24	4.5	1.5	2.4

(1) For specimens with 1/4 inch attachments.

(2) For specimens with 3/16 inch attachments.

TABLE 14. CYCLIC TEMPERATURE TEST

TEST NUMBER	AIR CYCLIC TEMPERATURE (°F + 10°F)		LOAD (LB)	TIME AT LOW TEMP (MINUTES)	TIME AT HIGH TEMP (MINUTES)	TRANSITION TIME LOW TO HIGH TEMP (MINUTES)	NUMBER OF TEMP CYCLES	TOTAL TEST TIME (HOURS)
	MIN	MAX						
4	-65	190	$\frac{735}{456}$	20*	20	12 or Less	5	4.33
5	160	245	$\frac{735}{456}$	20*	20	12 or Less	5	4.33
6	AMBIENT	245	$\frac{1470}{912}$	5	20	12 or Less	10	-

*First cycle at low temperature to be 60 minutes.

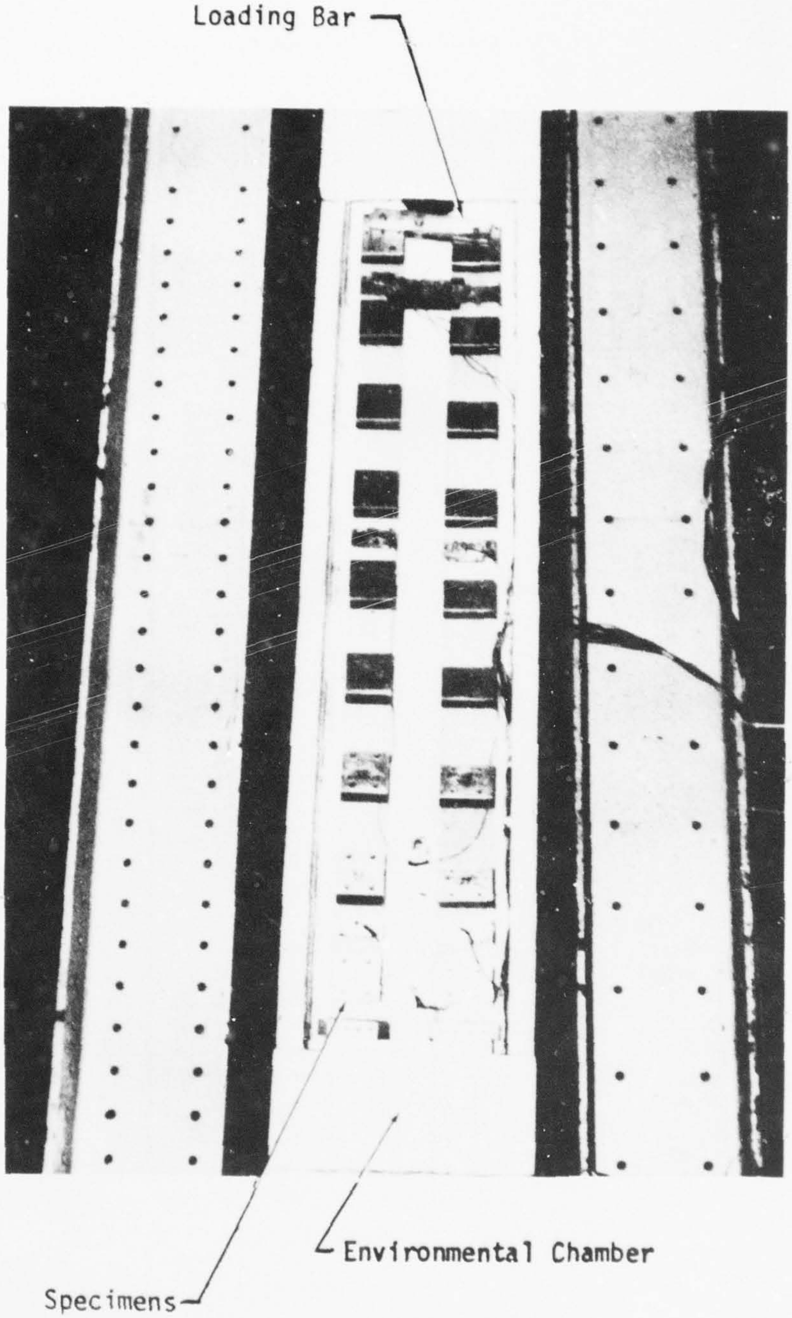


Figure 64. Specimen Assembly.

AD-A070 376 DOUGLAS AIRCRAFT CO LONG BEACH CALIF F/G 1/3
WINDSHIELD TECHNOLOGY DEMONSTRATOR PROGRAM-CANOPY DETAIL DESIGN--ETC(U)
SEP 78 M J COKER, J B HOFFMAN, J H LAWRENCE F33615-75-C-3105
UNCLASSIFIED MDC-J7176 AFFDL-TR-78-114 NL

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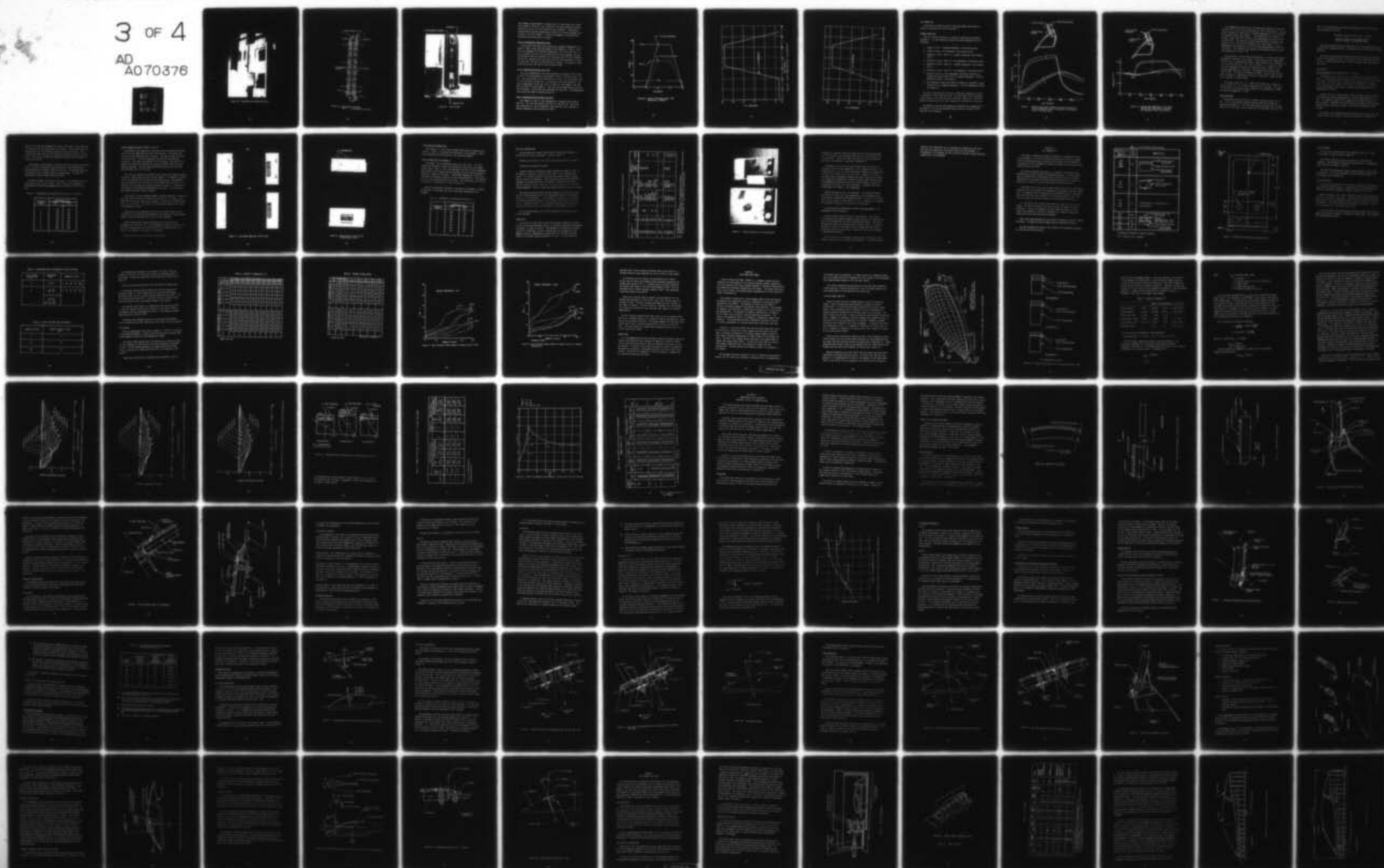
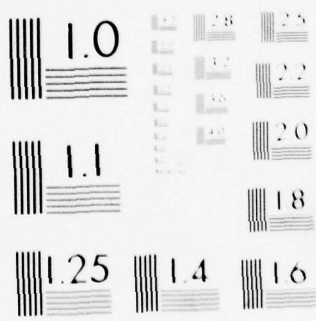




MICROCOPY RESOLUTION TEST CHART
NATIONAL BUREAU OF STANDARDS-1963-A

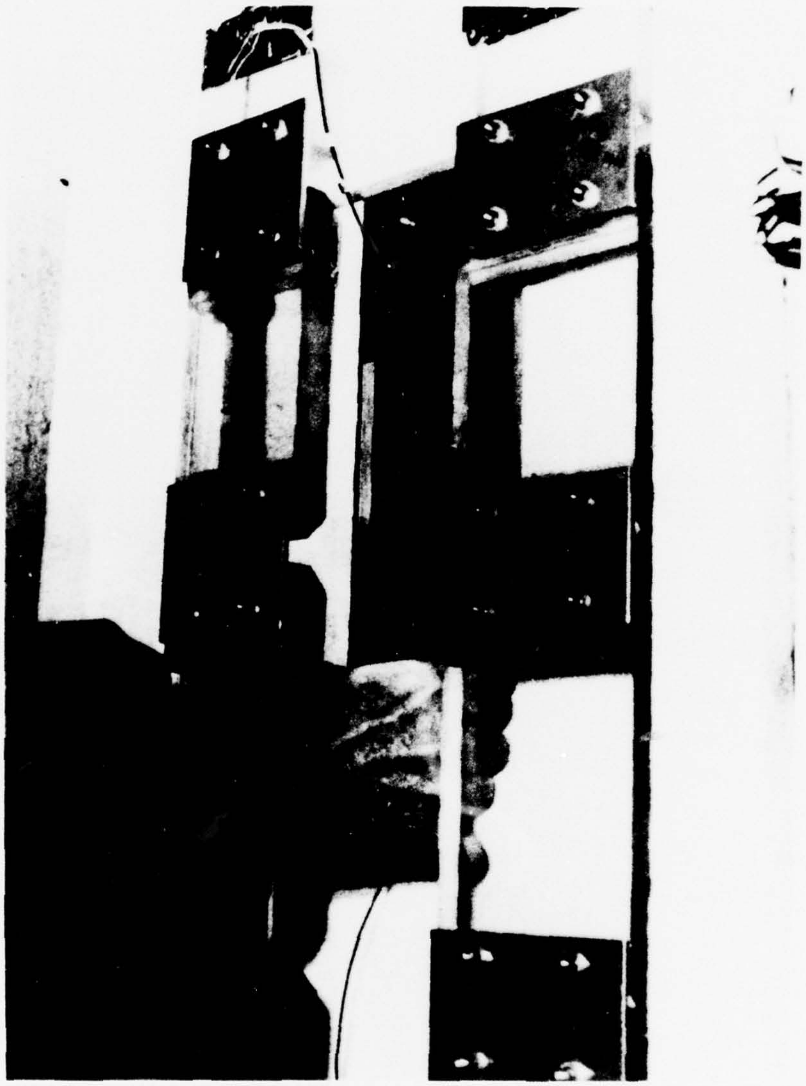


Figure 65. Specimens and Connecting Plates.

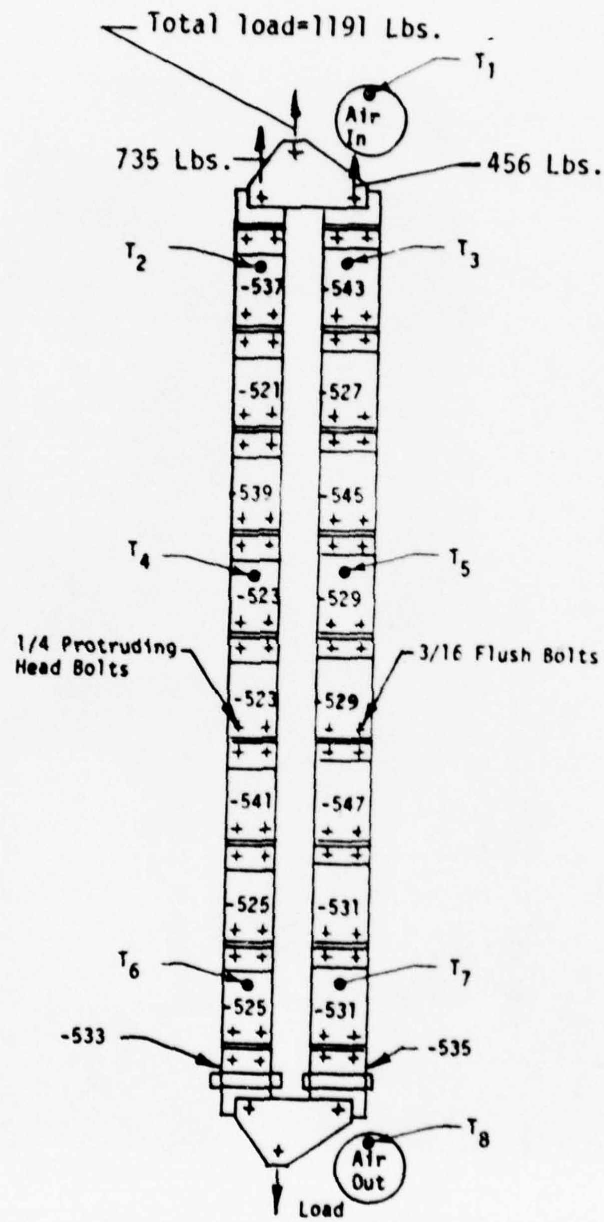


Figure 66. Specimen Test Assembly
(Note: T = Thermocouple Locations).

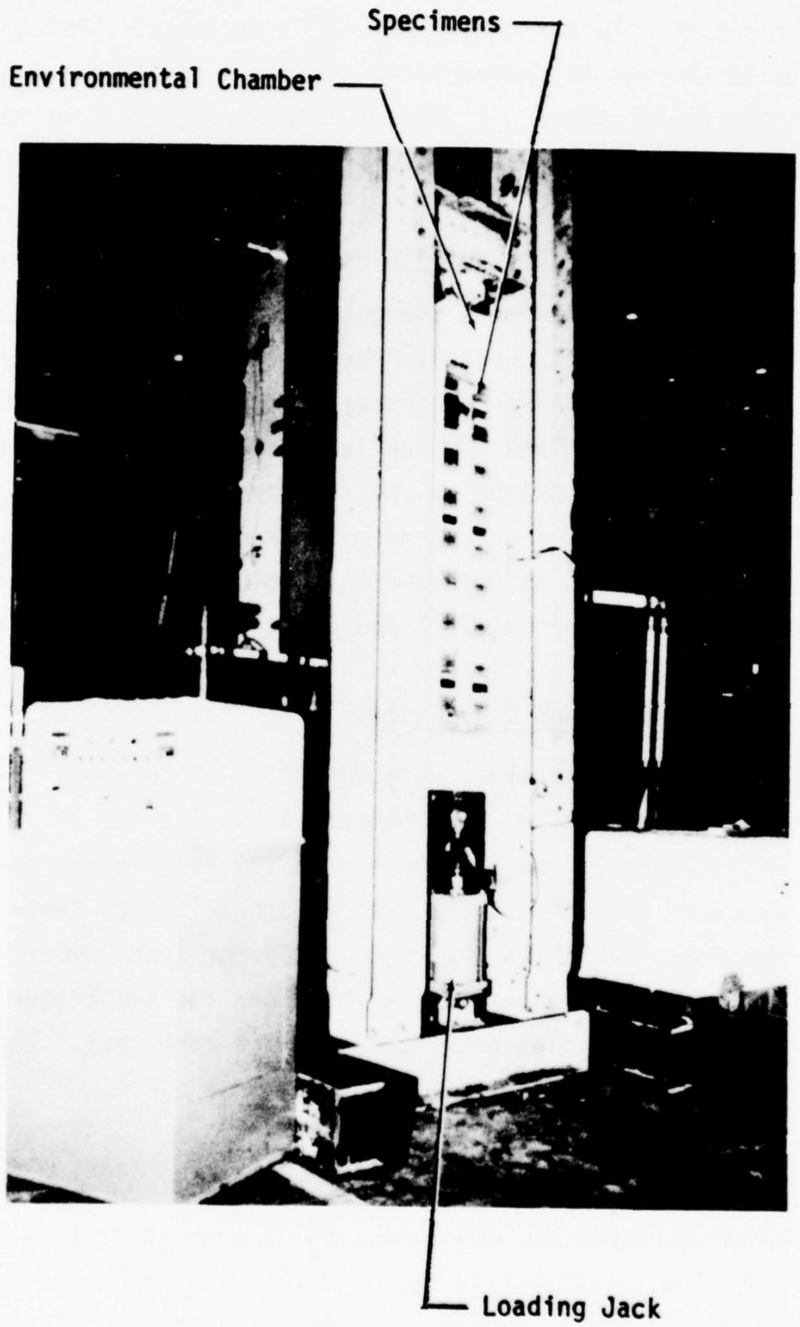


Figure 67. Test Fixture

test chamber, approximately 12 inches wide, 14 inches deep, and 7 feet high, shown in Figure 64, was installed on the test machine to provide ducting for hot or cold air around the test specimens. Temperatures were monitored through thermocouples attached to the test specimens and located in the test chamber. A Wilson Environmental Testing Machine delivered the hot and cold air supply.

Cyclic Load/Constant Temperature Test

Three cyclic load tests were conducted at constant temperatures, as shown in Table 13 and were designated as Test Numbers 1, 2 and 3. Each test was conducted at a different temperature, -5°F, 75°F and 160°F. A cyclic load of 0 to 1191 lbs. was applied to the test assembly, and was distributed as 735 lbs. maximum to the specimens with 1/4 inch fasteners and 456 lbs. maximum to the specimens with 3/16 inch fasteners, as shown in Figure 68. The cyclic time period was 6 minutes. The number of load cycles was 916 at -5°F, 874 at 75°F and 24 at 160°F.

Cyclic Temperature/Constant Load Test

Two cyclic temperature tests were conducted at constant load as shown in Table 14 and designated as Test Numbers 4 and 5. The total load, 1191 lbs., was distributed by the loading bar as 735 lbs. to the specimens with 1/4 inch fasteners and 456 lbs., to those with 3/16 inch fasteners. The cyclic temperature varied from -65°F to 190°F for Test Number 4 and from 160°F to 245°F for Test Number 5. The high and low temperatures were held for 20 minutes. Five cycles were conducted for each test. The temperature/time curves are shown in Figures 69 and 70.

Cyclic Temperature/Time Ultimate Load Test

Test Number 6, Table 14, was conducted at a constant total load of 2382 lbs. The load distribution was 1470 lbs. and 912 lbs. Ten cycles were conducted at a cyclic temperature that were held for 5 minutes at ambient temperature and 20 minutes at 245°F.

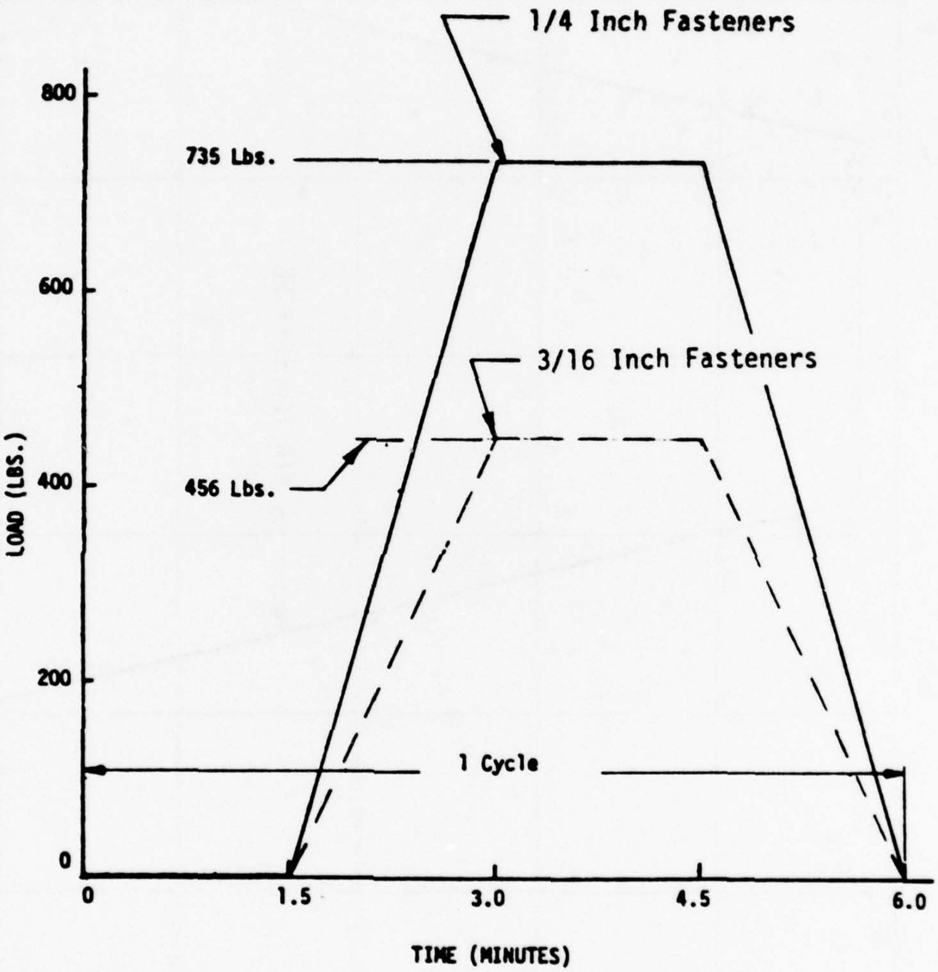


Figure 68. Typical Time Load Cycles, Test Conditions 1 through 3

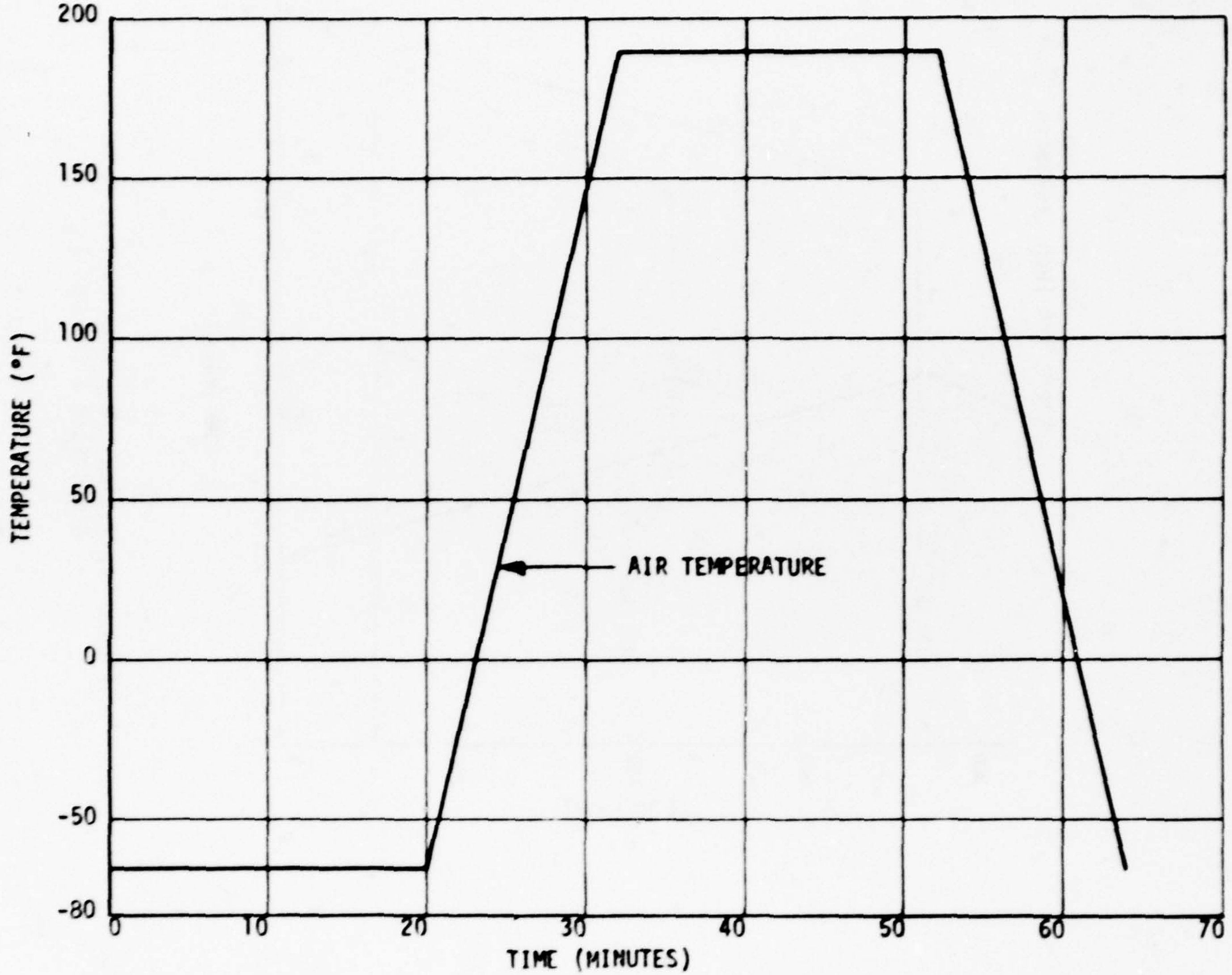


Figure 69. Typical Temperature Cycle, Test Condition 4

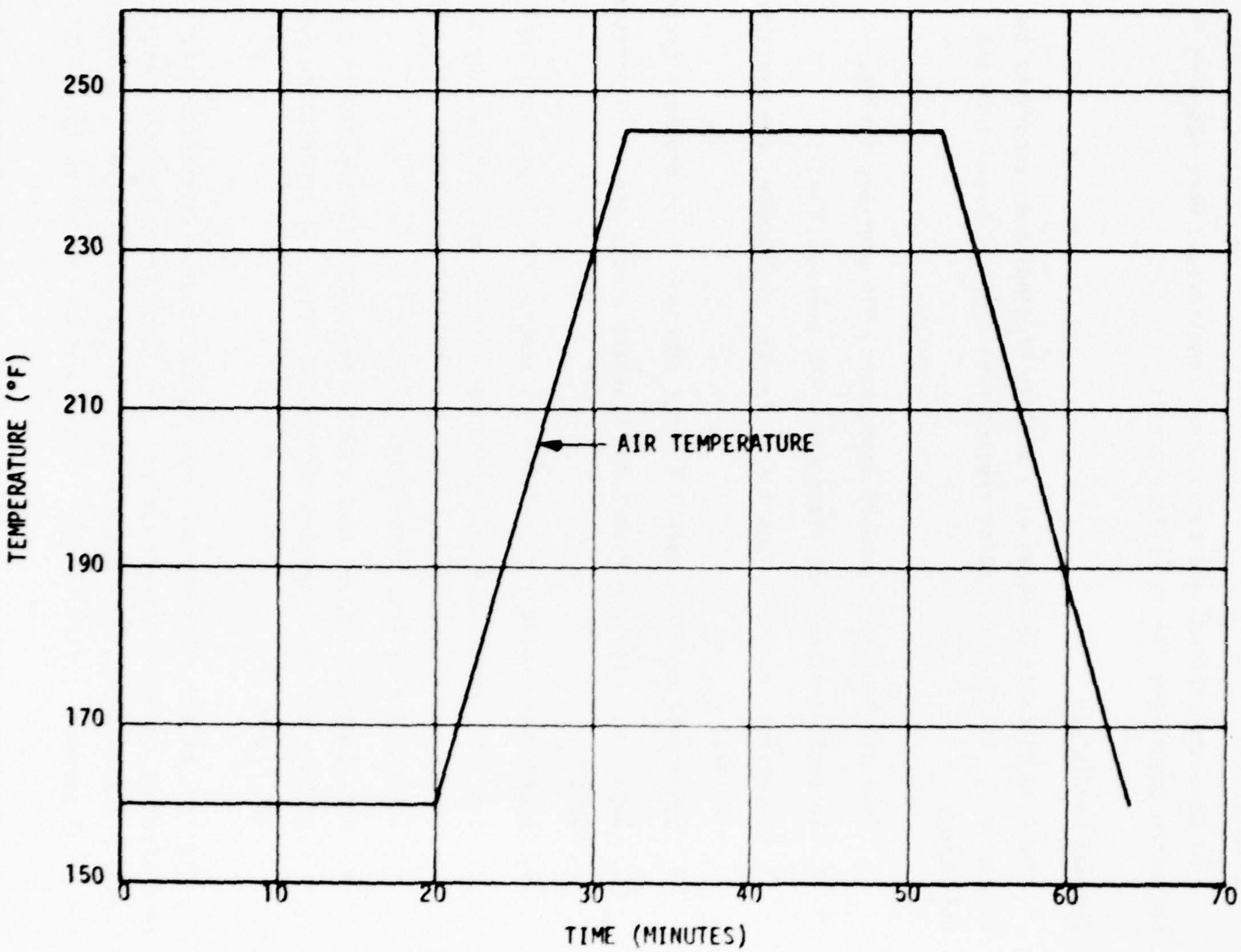


Figure 70. Typical Temperature Cycle, Test Condition 5

TEST CONDITIONS

The pressure, thermal and cyclic load requirements were based on operating conditions for the F-16 aircraft.

Thermal Conditions

The F-16 aircraft operates at a variety of conditions according to Reference 3. The many operating regimes have been condensed into the following:

1. Subsonic cruise - standard atmosphere, 916 pressure cycles.
2. Subsonic cruise - hot atmosphere, 102 pressure cycles.
3. Supersonic cruise - Mach 1.6 - standard atmosphere, 498 pressure cycles.
4. Supersonic cruise - Mach 1.6 - hot atmosphere, 55 pressure cycles.
5. Supersonic cruise - Mach 2.0 - standard atmosphere - 219 pressure cycles.
6. Supersonic cruise - Mach 2.0 - hot atmosphere, 24 pressure cycles.
7. Subsonic cruise in a cold atmosphere (65°F)soak) followed by a supersonic cruise of Mach 2.2 in a standard atmosphere - five (5) temperature cycles. See Figure 71.
8. Hot atmosphere ground soak (160°F) followed by a supersonic cruise of Mach 2.2 in a standard atmosphere - five (5) temperature cycles, see Figure 72.

The life of the aircraft was given as 8000 hours (Reference 3) which is equivalent to 7256 pressure cycles. For these tests the transparency/canopy pressure cycles were based on a 2000-hour design life. The total number of pressure cycles for one lifetime was calculated as 1814 cycles.

The thermal conditions were analyzed to determine the temperatures at the edge of a 5/8 inch polycarbonate canopy with an 0.08 acrylic face sheet and 0.10 interlayer.

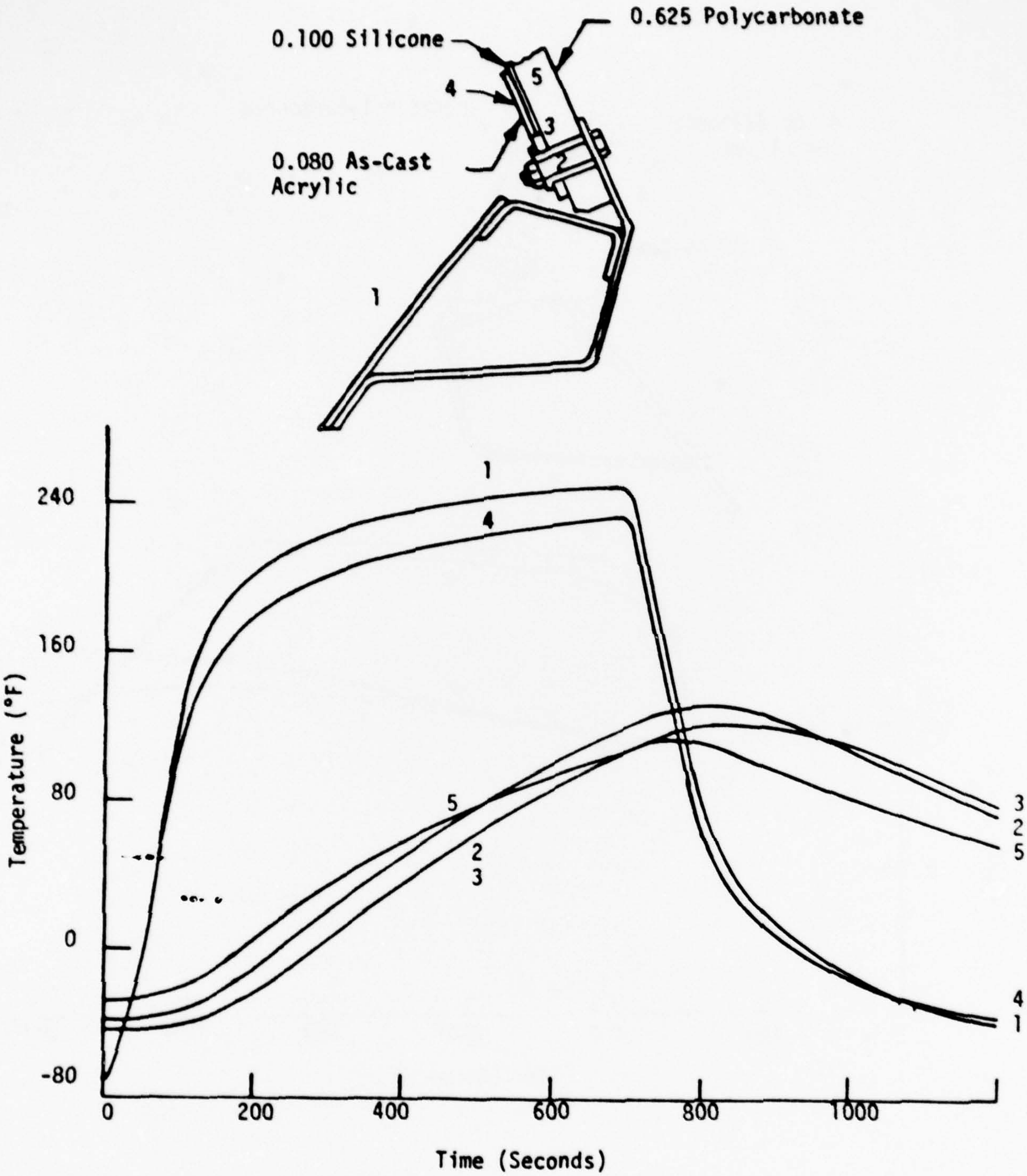


Figure 71. Average Canopy Edge Temperature During Acceleration From Cold Subsonic Cruise to Standard Day Supersonic Cruise, then Decelerate

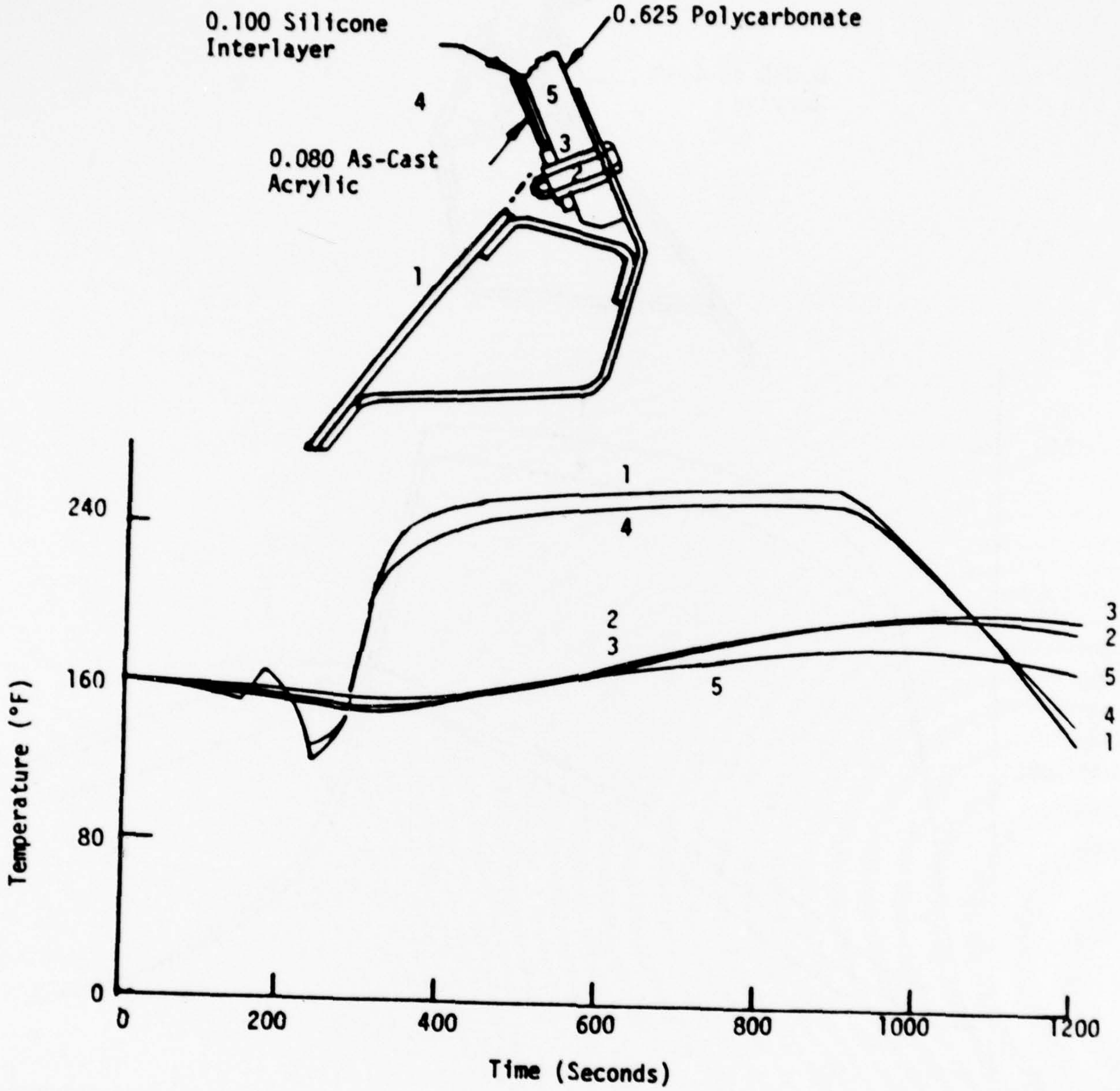


Figure 72. Average Edge Temperature During Climb from Hot Day Ground Soak to Standard Day Supersonic Cruise, then Descent

The temperatures which occur at the edge during Condition 1 were calculated to be -5°F on the polycarbonate and -15°F on the acrylic. The condition selected was specified as a constant temperature of -5 ± 10°F and 916 pressure cycles, Test Number 1. The temperatures for Conditions 2 through 5 range from 18°F to 93°F on the polycarbonate and 49°F to 115°F on the acrylic. The condition selected were defined as a constant temperature of 75 ± 10°F and 874 pressure cycles, Test Number 2. The temperatures for Condition 6 are 110°F on the polycarbonate and 160°F on the acrylic. This condition was defined as 24 pressure cycles at a constant temperature of 160 + 10°F.

Two conditions, 7 and 8, were selected which will produce the largest temperature transients in the canopy. Since these are not probable conditions, only five temperature cycles of each condition were selected to be run at constant pressure. The first, Condition 7, is a subsonic cruise in a cold atmosphere followed by a supersonic cruise of Mach 2.2 in a standard atmosphere for 10 minutes. The temperature extreme is from -65 to 190°F. The cycle was defined as follows: soak at -65°F for one hour, increase air temperature as quickly as possible to 190°F and hold for 20 minutes, and then repeat.

The second, Condition 8, is a 160°F soak followed by a supersonic cruise at Mach 2.2, in a standard atmosphere for 10 minutes. The cycle is the same as the previous cycle, except the low temperature is 160°F and the high temperature is 245°F.

Load Conditions

The applied load was based on an internal cockpit pressure of 13.2 psi. This is the ultimate internal pressure condition assumed for a canopy per Reference 15. Two load values were calculated, one for specimens with 1/4 inch bolts and the second for the specimens with 3/16 inch

bolts. The load that was applied to the specimens with 1/4 inch bolts was calculated as follows:

$$(13.2 \text{ PSI}) \times (15.9 \text{ IN.}) \times (3.50 \text{ IN.}) = 735 \text{ LB.}$$

where 15.9 inches is the canopy radius
and 3.50 inches is the specimen width.

The total load applied to the specimens with 1/4 inch bolts was 735 pounds or 367.5 pounds per bolt. The load per inch of width was 210 pounds per inch.

The total load applied to the specimens with 3/16 inch bolts was 456 pounds or 228 pounds per bolt. The load per inch of width was 114 pounds per inch. These loads were used for Tests 1 through 5 and were doubled for Test 6.

TEST RESULTS

Life Cycle One (Cyclic Load Tests No. 1, 2 and 3)

Test Number 1 was a simulation of a subsonic cruise, standard atmosphere. The estimated time to complete, based on six minutes per cycle was 91.6 hours. The actual time to complete was approximately 110 hours. A total test load of 1191 pounds (735 + 456) was achieved. The capacity of the cooling agent (CO_2) tanks, permitted an uninterrupted run of approximately 24 hours.

The test was run 16 hours per day, 5 days per week, and was monitored. The coolant was supplied automatically to keep the temperature at -5° $\pm$ 10°F. An automatic shutoff valve stopped the test when the tanks ran out of coolant. Inspection of the specimens was accomplished three times per day at each eight-hour interval. This test was completed without incident.

Test Number 2 was delayed because the bushings pulled out of the -521 configuration at cycle Number 325 and caused the time to complete the

test to slip from the estimated 87.4 hours to 110 hours. This occurred at the end of the specimen without the retainer. The bushings were reinstalled using one inch diameter washers under the bolt heads. This procedure was duplicated for the remaining specimens utilizing 1/4 inch bolts. The test was resumed and completed without further incident for the full 874 cycles.

Test Number 3 precipitated a failure to the -533 specimen on the first cycle. The holes had been drilled improperly on the non-test end of the specimen which produced a shear-out failure. The specimen could not be repaired and was replaced with an aluminum spacer. The test was completed as planned without further incident.

Electrical heaters were used for this test. The selected load, temperature and cycles were achieved. Table 15 shows the temperature distribution within the environmental box for Cycles 1 and 10 of this test.

TABLE 15. TEMPERATURE DISTRIBUTION FOR TEST NO. 3, LIFE CYCLE 1

THERMOCOUPLE NUMBER	TEMPERATURE (°F)	
	CYCLE 1	CYCLE 10
1	164	167
2	160	163
3	161	164
4	154	160
5	162	166
6	153	157
7	150	156
8	160	163

Cyclic Temperature Tests (Numbers 4 and 5)

Test Number 4 was completed as required with the following variations. The initial low temperature was held for 60 minutes. The target time for cold to hot temperature was 12 minutes. The actual time varied from 13 to 25 minutes. The high temperature was held for the specified 20 minutes. Cool down time required 13 to 18 minutes. The load of 1191 pounds was accomplished although the pressure regulator had to be continually adjusted. The bolt loads were 367.5 pounds per 1/4 inch bolt and 228 pounds per 3/16 inch bolt.

This test was stopped on the fourth cycle because the loading cylinder ran out of stroke before maximum load was reached. The combined stretching of all the specimens was the cause of the bottoming out. No permanent set was observed in the specimens after the completion of the test. The specimens had been marked to measure permanent elongation. Adjustment of the hydraulic jack permitted continuation of the test. The five cycles were completed without further incident.

Test number 5 caused permanent damage to several specimens as described. The time to heat from 160°F to 245°F varied from 5 to 25 minutes. The 245°F was held for the specified 20 minutes. Time to cool to 160°F varied from 13 to 18 minutes. The load of 1191 pounds was accomplished for the five temperature cycles.

Observation at the completion of five cycles showed bubbles in the interlayer of the specimens fabricated with the Sierracin S-130 interlayer. Figure 73 shows typical bubbling in two of these specimens.

Further observations revealed delamination in the bolt area of Specimens -525, -529 and -531. This delamination, shown in Figure 74 for the -531 configuration, occurred at the end of the specimen where the acrylic ply extended to the edge of the specimen.

No crazing was observed on any of the specimens.

-537

-543

Figure 73. Interlayer Bubbling (-537 & -543)

Figure 74. Delamination of Acrylic Ply at Bolt Holes (-531)

Life Cycle Two Through Four

Test Numbers 1, 2 and 3 were repeated three times to complete Life Cycles 2, 3 and 4. No changes were observed in the specimens. These tests completed 7256 load cycles on the specimen assembly.

Cyclic Temperature Test Number 5

Test Number 6 was accomplished after Life Cycle Four. This test loaded the specimens to twice the operational load at the high temperatures seen in Test 5. Thus, the specimens with the 1/4 inch attachments were loaded to 1470 pounds and the specimens with the 3/16 inch attachments were loaded to 912 pounds. The maximum temperature of 245°F was limited by the thermal capability of the environmental box. The required 24 cycles were completed without incident. The temperature distribution for the eight thermocouples is shown in Table 16.

Post-test observations revealed an increase in the number of bubbles in Specimens -527 through -547 but no additional delamination was observed.

TABLE 16. TEMPERATURE DISTRIBUTION FOR TEST NO. 6

THERMOCOUPLE NUMBER	TEMPERATURE (°F) CYCLE 1	TEMPERATURE (°F) CYCLE 10
1	243	238
2	241	242
3	246	247
4	240	243
5	235	239
6	233	239
7	233	237
8	239	237

POST-TEST OBSERVATIONS

The specimens were removed from the test fixture and inspected. A description of the test specimens is given in Table 17.

Compressive yielding of the acrylic ply occurred under the washers.

No measurable creep had occurred.

Surface crazing was observed on many specimens as noted in Table 17. The degree of crazing is defined as very light, light, moderate and severe. All crazing occurred on the specimen surface under the plates used to connect the specimens. Those specimens that were installed with urethane tape between the plate and specimen, exhibited less crazing than the specimens without the tape. It is not known at what cycle the crazing appeared. Figure 75 shows the surface crazing in the -521 specimen.

The surface crazing existing on these specimens is believed to have resulted from contamination of the polycarbonate combined with tension stress and high temperature. The asymmetrical loading of the specimens induced maximum tension stress on the specimen surface that interfaces with the connecting plates. The calculated tension stress on the connecting plate side of the -521 specimen was approximately 3500 psi.

No bushing deformation or bolt hole elongation was visible in any of the specimens.

CONCLUSIONS

The results of these tests verify the structural integrity and durability for the edge attachment area of a laminated windshield configuration as defined in Figures 59 and 60. The adequacy of the design was substantiated for an 8000-hour aircraft lifetime which is equivalent to 7256 pressure cycles at temperatures ranging from -65°F to 245°F and loads based on an ultimate canopy pressure of 13.2 psi. The load condition

TABLE 17. POST-TEST SPECIMEN DESCRIPTION

CONFIGURATION OF Z5943266	MFG	TAPE UNDER BOLTS	CRAZING		COMPRESSIVE YIELDING UNDER WASHERS	REMARKS (1)
			NON-RETAINER END	RETAINER END		
-521	SK		Severe	Moderate	Yes	
-523	SK		Severe	Severe	Yes	
-523	SWU	Yes	No	No	Yes	
-525	SK	Yes	Moderate	Light	Yes	
-525	SWU		Severe	Severe	Yes	(2)
-527	SK		Moderate	Moderate		
-529	SK		Moderate	Moderate		
-529	SWU	Yes	Very Light	No		(2)
-531	SK		Moderate	Moderate		
-531	SWU	Yes	Very Light	Very Light		(2)
-533	TEX		-	-		
-535	TEX		No	No		
-537	SK		Severe	Moderate	Yes	(3)
-539	SK		Severe	Moderate	Yes	(3)
-541	SK		Severe	Moderate	Yes	(3)
-543	SK		Light	Light		(3)
-545	SK		Light	No		(3)
-547	SK		Light	Light		(3)

(1) No measurable creep occurred.
(2) Delamination occurred in bolt area at the end without the retainer.
(3) Bubbles observed in S-130 interlayer at completion of first life cycle, high temperature test. The number of bubbles increased with successive high temperature tests.

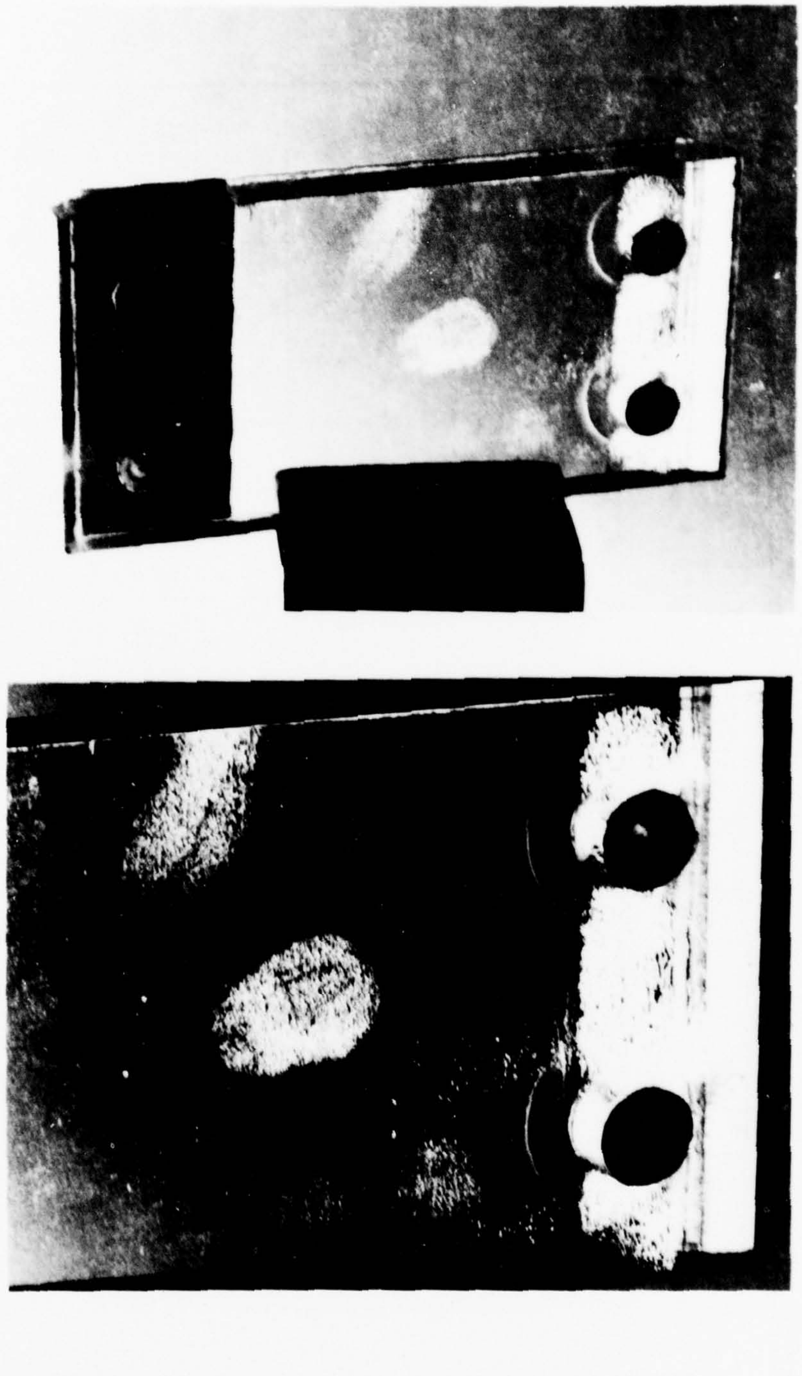


Figure 75. Surface Crazing (-521 Configuration)

resulted in a specimen load of 210 pounds per inch for configurations with 1/4 inch bolts, and 114 pounds per inch for configurations with 3/16 inch bolts. These tests were similar to the tests conducted on monolithic polycarbonate as reported in Reference 22.

Delamination did occur at the non-retainer end of one group of specimens during the high temperature, cyclic temperature tests. This delamination was a result of high temperature combined with compression induced in the acrylic ply by the bolt head. Asymmetrical loading by the single shear arrangement induced a load that caused bushing pullout. This problem was eliminated by installing washers under the bolt heads. High temperature combined with the bolt load resulted in compressive yielding of the acrylic ply under the washers. Extending the acrylic to the edge of the canopy is not recommended.

Bubbles occurred in the S-130 interlayer. The bubbling occurred during the Life Cycle One, high temperature, cyclic temperature test and grew with subsequent high temperature tests. The use of this interlayer as fabricated for these specimens is not recommended for laminated transparencies expected to experience high temperatures.

The specimens had been marked to permit creep measurements. No measurable creep was observed.

Surface crazing occurred in many of the specimens. The crazing occurred under the aluminum connecting plates and was less severe at the end of the specimen that utilized the retainer and spacer. The use of urethane tape under the connecting plates reduced crazing. As noted in Reference 22, the use of an agent such as urethane tape along with a protective coating can significantly reduce the effects of environment on polycarbonate surfaces.

The test assembly was designed to load and test 18 specimens simultaneously under high, low and ambient temperatures. This method assured

identical test conditions for all specimens and reduced the test time that would be required for individual testing of each specimen. A disadvantage of this method was the difficulty involved in making periodic inspections of the specimens.

SECTION IX
SALT ABRADER TEST

The design of supersonic aircraft windshields is influenced by abrasion resistance. The transparency must be able to withstand the "impact abrasion" encountered when an aircraft flies through a cloud containing ice crystals or an environment of airborne dust and still provide an undistorted view for the pilot.

This section presents the results of a series of salt abrader tests performed on aircraft windshield transparent materials. Each test specimen was impacted with a controlled blast of salt particles at selected temperatures to simulate an aircraft windshield encounter with minute ice particles in clouds or with dust when flying close to the ground.

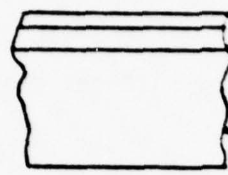
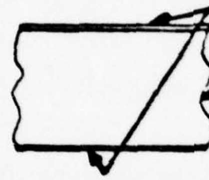
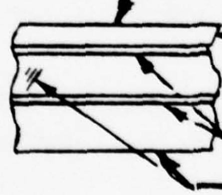
The purpose of this test was to provide data to be used as a qualitative comparison for an assortment of representative transparent windshield specimens. The measure of this comparison was the percent of haze in the specimens after they have been subjected to a controlled number of salt blast cycles at identical near-sonic velocities and selected temperatures.

Six transparent test specimen configurations, shown in Table 18, were tested. The face-ply materials and the cross section composition are identified. Four of the specimens had been tested previously in a wind tunnel test program as reported in Reference 25. Figure 76 shows the configuration plan form and the area impacted.

These tests were performed at the Sierracin Corporation facility, Sylmar, California. A Douglas Aircraft Company engineer witnessed the tests.

The test procedure was based on salt abrader tests performed previously by Sierracin and PPG Industries.

TABLE 18. TEST SPECIMEN CONFIGURATIONS

CONFIGURATION[1] DAC S/N VENDOR S/N[2]	PRIOR WIND TUNNEL TEST	CROSS SECTION
-501 **SK06** **SN003**	No	 0.080 As Cast Acrylic (MIL-P-8184); 0.100 Silicone (S-100); 0.625 Polycarbonate (MIL-P-83310)
-505 SK11 SN002	No	 Sierraclad; 0.625 Polycarbonate (MIL-P-83310)
-517 SK16 SN002	Yes	0.900 Monolithic Stretched Acrylic (MIL-P-25690)
-519 **GY01**	Yes	 0.100 Acrylic (For -519) (MIL-P-8184); Urethane (GAC-590-55)(For -521) (GAC-590-65)(For -523); 0.040 Urethane (F5X-3A); 0.250 Polycarbonate (MIL-P-83310)
-521 **GY02**	Yes	
-523 **GY03**	Yes	

[1]Configuration Numbers of Drawing Z5943260.

[2]SK = Sierracin, GY = Goodyear.

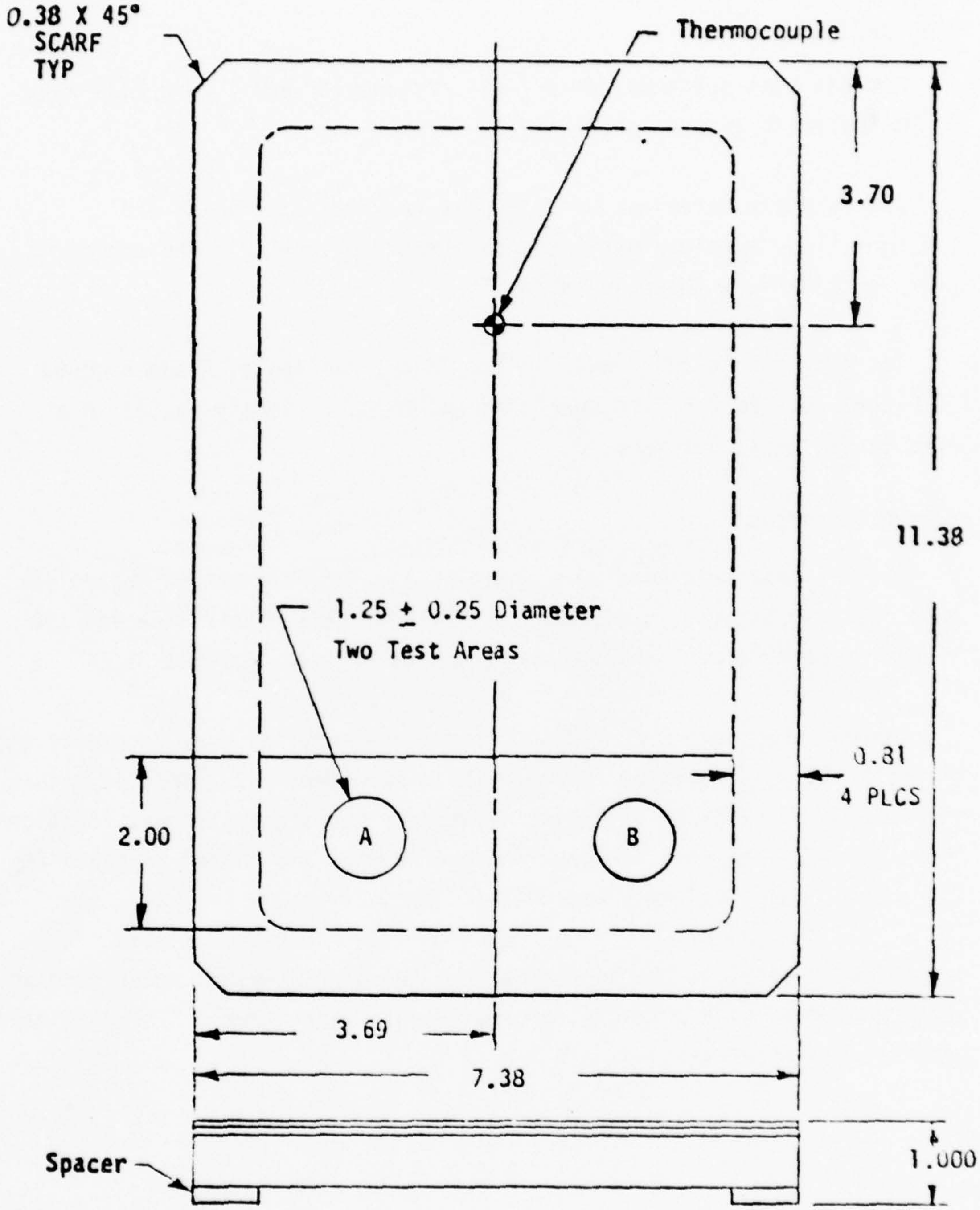


Figure 76. Windshield Test Specimen, General Outline.

TEST SPECIMENS

The six test specimens were flat, rectangular and 1.00 x 7.38 x 11.38 inches in size as shown in Figure 76.

The face ply materials were as-cast acrylic, Sierraclad and urethane. One specimen was uncoated stretched acrylic. The specimen cross sections are shown in Table 18.

The area salt blasted was limited to the two spots, A and B noted in Figure 76. Four of the specimens had been previously tested in a wind tunnel test, Reference 25.

TEST DESCRIPTION

The six test specimens were individually impacted at the controlled spot on their exterior surface. Intermittent salt blasts impacted the specimens under three temperature ranges as defined in Table 19.

Control specimens (MIL-P-25690, stretched acrylic) were furnished by Sierracin to calibrate the system. These specimens were impacted before each series of tests to determine whether or not the system was functioning properly. The control specimen requirements are listed in Table 20 for a one second "on", one second "off" cycle.

Prior to testing, the specimens were thoroughly washed under running water and dried with a Kaydry, or equivalent, paper towel. The specimens were cleaned after testing.

TABLE 19. TEMPERATURE/CYCLE REQUIREMENTS FOR EACH SPECIMEN

TEST LOCATION (Figure 76)	TEMPERATURE (°F)	NUMBER OF CYCLES
A	75 ± 15	5 10 20 50 100
B	$265\,^{+0}_{-75}$	5 10 20 50 100
	$200\,^{+0}_{-50}$ FOR SK16 ONLY	

TABLE 20. CONTROL SPECIMEN HAZE REQUIREMENTS

NUMBER OF CYCLES	PERCENT INCREASE IN HAZE ±1%
5	9
10	13
20	15

The specimens were supported for testing in a fixture that was adjusted so that the salt blast would strike the specimen normal to the surface. A mask was installed to limit abrasion to area A and B, Figure 76.

The salt used for abrading was dried and sifted to remove lumps.

A sand blast gun was used to direct the salt blast against the specimen surface. A timer control was set to operate the abrader one second "off" (no air, no salt) and one second "on" per cycle. The air vessel pressure was 40 psi. The tip of the gun was eight inches from the specimen surface and mounted normal to the surface.

The specimens were heated to a temperature above those specified in Table 19, removed from the oven, mounted in the support fixture, and impacted per the requirements of Table 19. The maximum specimen temperature was limited to 295°F.

A Gardener type hazemeter was used to record haze measurements prior to testing and at the completion of each level of cyclic abrasion.

TEST RESULTS

The test temperatures are listed in Table 21. Table 22 is a record of the haze measurements. Figures 77 and 78 are plots of percent haze versus number of cycles for each temperature regime.

The range of haze varied from 6 to 8 percent at 5 cycles and from 9.5 to 19 percent at 100 cycles for the tests conducted on specimens at 74°F. For the heated specimens, the range of haze varied from 6 to 12.5 percent at 5 cycles and from 14.5 to 27.5 percent at 100 cycles.

These tests indicate that stretched acrylic materials, such as

TABLE 21. RECORD OF TEMPERATURE (°F)

	TEST NO.	SERIAL NO.	5 CYCLES		10 CYCLES		20 CYCLES		50 CYCLES		100 CYCLES	
			PRE-TEST	POST TEST	PRE-TEST	POST TEST	PRE-TEST	POST TEST	PRE-TEST	POST TEST	PRE-TEST	POST TEST
LOCATION A (75°F)	01	SK06	74	74	74	74	74	74	74	74	74	74
	02	SK11										
	03	SK16										
	04	GY01										
	05	GY02										
	06	GY03	74	74	74	74	74	74	74	74	74	74
LOCATION B (265°F)*	07	SK06	250	246	250	245	250	245	250	225	250	206
	08	SK11	250	242	250	245	250	244	250	220	250	210
	09	SK16	200	187	200	185	200	189	200	188	200	172
	10	GY01	250	245	250	244	250	246	250	218	250	205
	11	GY02	250	242	250	246	250	246	250	218	250	208
	12	GY03	250	238	250	246	250	243	250	215	250	203

*200°F for SK16

TABLE 22. RECORD OF HAZE VALUES

	TEST NO.	SERIAL NO.	PRE-TEST	5 CYCLES	10 CYCLES	20 CYCLES	50 CYCLES	100 CYCLES
LOCATION A (75°F)	01	SK06	5.5	6.1	6.9	7.3	10.4	11.7
	02	SK11	5.3	5.9	8.0	9.7	11.3	12.6
	03	SK16	7.1	7.6	9.8	16.0	18.4	17.8
	04	GY01	5.1	5.8	7.7	9.0	15.8	19.2
	05	GY02	5.9	5.9	5.9	6.1	7.8	9.5
	06	GY03	5.7	7.0	8.7	10.0	15.2	16.8
LOCATION B (265°F)*	07	SK06	5.5	5.8	7.9	8.8	12.2	16.3
	08	SK11	5.3	7.1	7.9	9.2	11.5	15.9
	09	SK16	7.1	10.6	15.9	22.5	24.8	26.9
	10	GY01	5.1	6.7	11.7	11.2	13.3	14.5
	11	GY02	5.9	8.3	8.1	10.4	14.5	17.2
	12	GY03	5.7	12.3	14.7	17.3	25.5	27.4

(Readings in Percents.)

*200°F for SK16

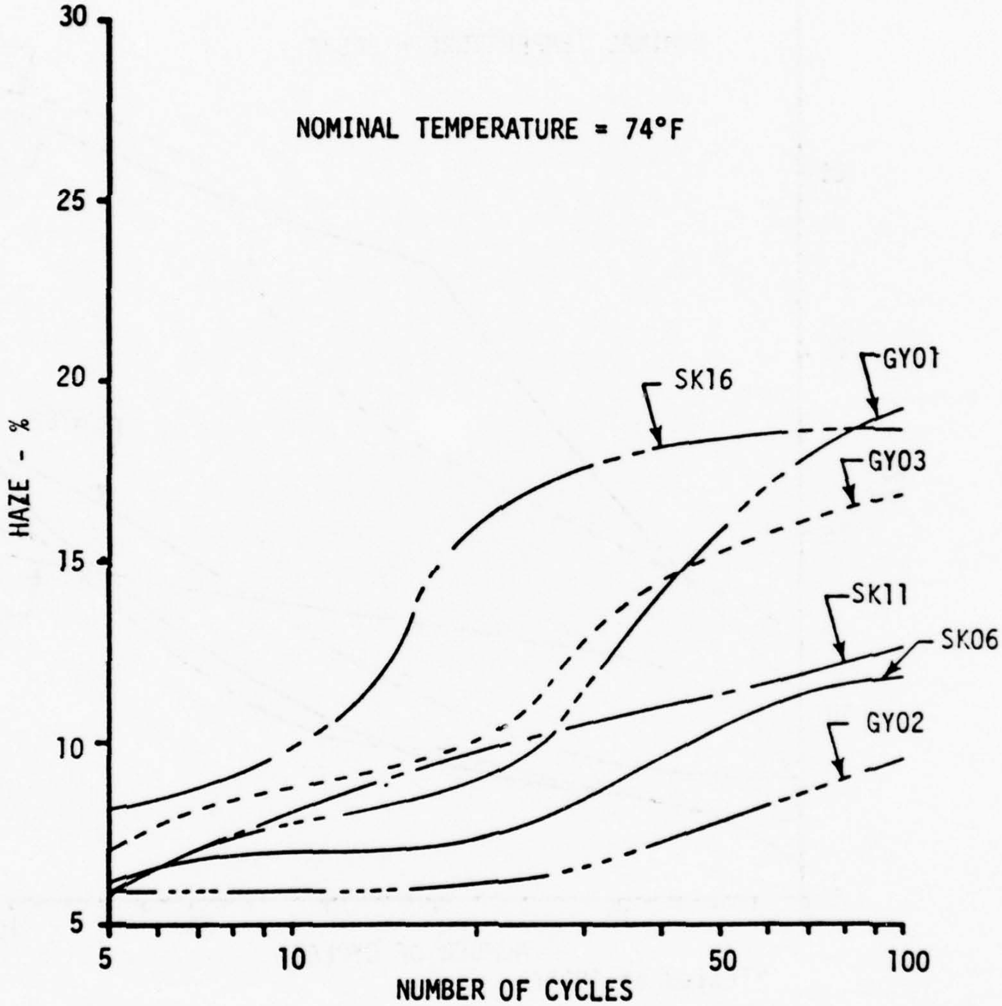


Figure 77. Haze (Percent) Versus Number of Impact Cycles at 74°F.

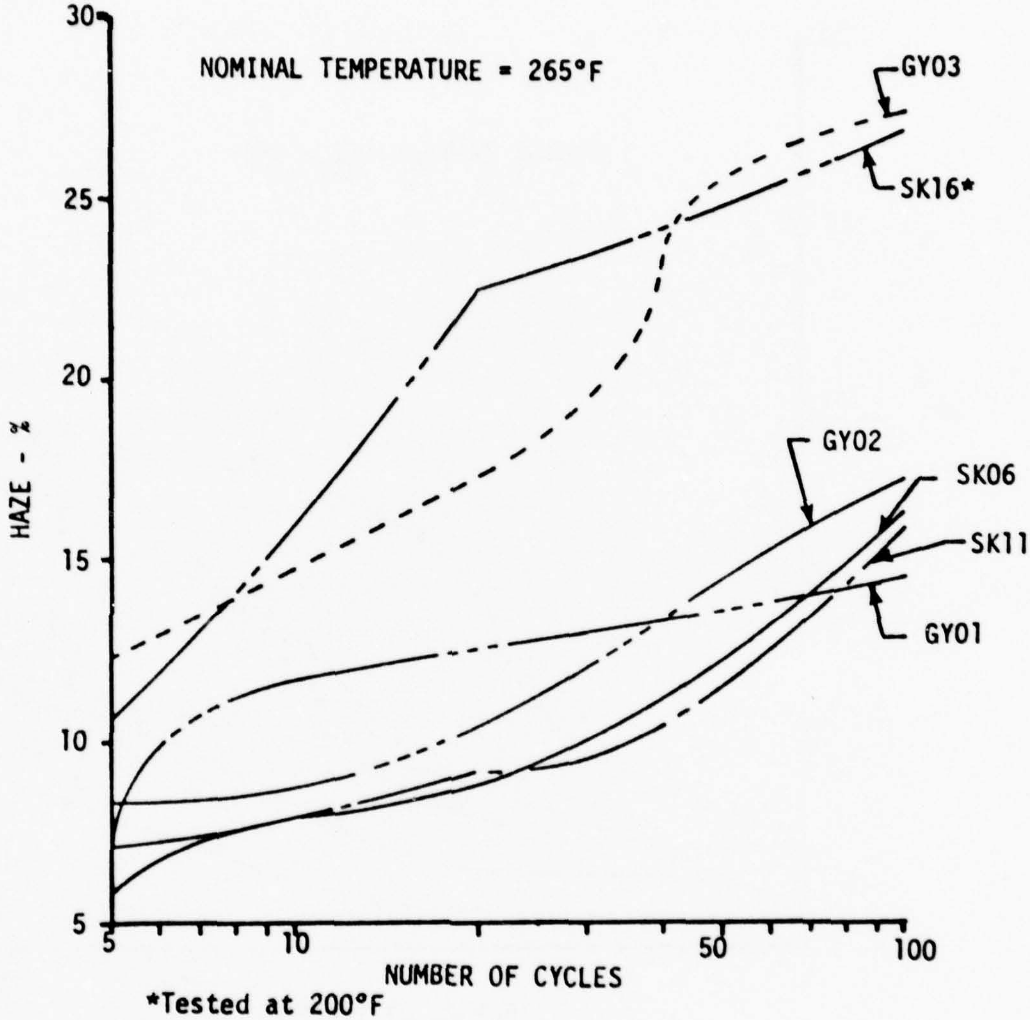


Figure 78. Haze (Percent) Versus Number of Impact Cycles at Elevated Temperatures.

specimen SK16, are more prone to abrasion than as-cast acrylic or urethane materials when subjected to up to 40 cycles of salt impact.

At 100 cycles of salt impact, the stretched acrylic (SK16), as-cast acrylic (GY01) and urethane (GY03) specimens registered the highest haze values at 74°F. A urethane face ply (GY03) registered the highest haze value at 100 cycles when heated to 265°F, and the stretched acrylic (SK16) specimen, heated to 200°F, also registered a high haze value at 100 cycles.

These tests indicate that any plastic face ply material will be abraded when subjected to a salt blast; thus, this abrasion causes an increase in haze. The haze increases nonlinearly with an increase in the number of impacts. With one exception, Specimen GY01, the maximum haze values were recorded for the specimens when impacted at elevated temperatures.

At 74°F the haze values varied from a low of 9.5% for Specimen GY02, which has a urethane face ply, to a high of 19.2% for Specimen GY01, which has an as-cast acrylic face ply. At elevated temperatures the haze values varied from a low of 14.5% for Specimen GY01 to a high of 27.4% for GY03. The haze reading for Specimen SK16 (stretched acrylic) was 17.8% at 74°F and 26.9% at 200°F.

CONCLUSIONS

It is recommended that further studies be made to correlate these test results with windshield haze readings per number of flights for actual aircraft operating conditions in order to determine the accuracy of the test method. A variety of tests are being used by industry to determine the abrasion resistance of transparent materials used on aircraft windshields. The ASTM F7.08 committee is trying to develop a standard test for determining abrasion resistance.

SECTION X
BIRD IMPACT MATH MODEL

The Bird Impact Math Model (IMPACT) is a computer program designed especially for the purpose of calculating transient dynamic responses of aircraft windshield and canopy systems composed of laminated transparencies and supporting structures subjected to bird impact. The theory and applications, user's manual, and program manual documentation for this report are found in Reference 26.

The program is based upon a finite element model of the multilayered transparency and supporting structure, subjected to time-varying loads representing bird impact. The equation of motion for the model is derived, considering geometric and material nonlinearities. The approach to geometric nonlinearities is based upon the method of fictitious forces and deformations. The approach to material nonlinearities is based on the Von Mises yield criterion, and the Pandtl-Reuss equations. The scope of the computing effort is minimized by introducing a modal transformation. The transformed nonlinear differential equation of motion is solved incrementally in time and iteratively within each time step.

Initially, it was anticipated that the bird impact math model would be utilized as a design tool, after appropriate correlation with test data, to analyze the potential F-16 canopy changes needed to design a bird impact resistant transparency for impact levels of 350 through 562 knots. Analysis of ten canopy design cases were to be accomplished to support the canopy design effort. To obtain the test data for correlation between program and test, the Canopy Bird Impact Test was accomplished. The test plan for this series of tests is an appendix of this report.

The attempted nonlinear analyses of the F-16 canopy were unsuccessful. Reasons for the failure are discussed in Reference 26 and recommendations

for further study are presented. A linear analysis was computed successfully and showed qualitative agreement with test results, but displacements were underestimated because of the nonlinear effects.

This section describes the application of the math model program to linear analyses of three configurations of the F-16 three-layered clear view canopy design for the 350 knot bird impact case.

F-16 MATH MODEL ANALYSIS

Figure 79 shows the finite element model for the F-16 clear view canopy. Three configurations were analyzed in support of the DAC design effort. All had the same frame structure, but the joint coordinates in the transparency were re-computed for each configuration to produce different thicknesses of the three layers in the laminate. The inner surface did not move. Figure 80 shows the three thickness configurations. Symmetry of structure, loads and response was assumed to exist, so that only half of the structure had to be modeled. Symmetric boundary constraints were imposed at the plane of symmetry. In addition, the model was constrained in the X, Y, Z degrees of freedom at the hinge point, in the Y, Z degrees of freedom at the latches and in the Y degree of freedom at the forward bearing point.

Table 23 gives the material properties used in the transparency layers. These represent low strain rate design properties at room temperature and are taken from data available during the preparation of Reference 18. These properties were actually input to the program, but for linear analysis the program uses only the Elastic Modulus (E) and Poisson's Ratio (γ).

Material properties vary with strain rate and strain rate varies with time during the canopy birdstrike event. Low strain rate data was used because it is representative for most of the elements in the canopy model. Areas of high strain rate are expected to be localized and to be moving

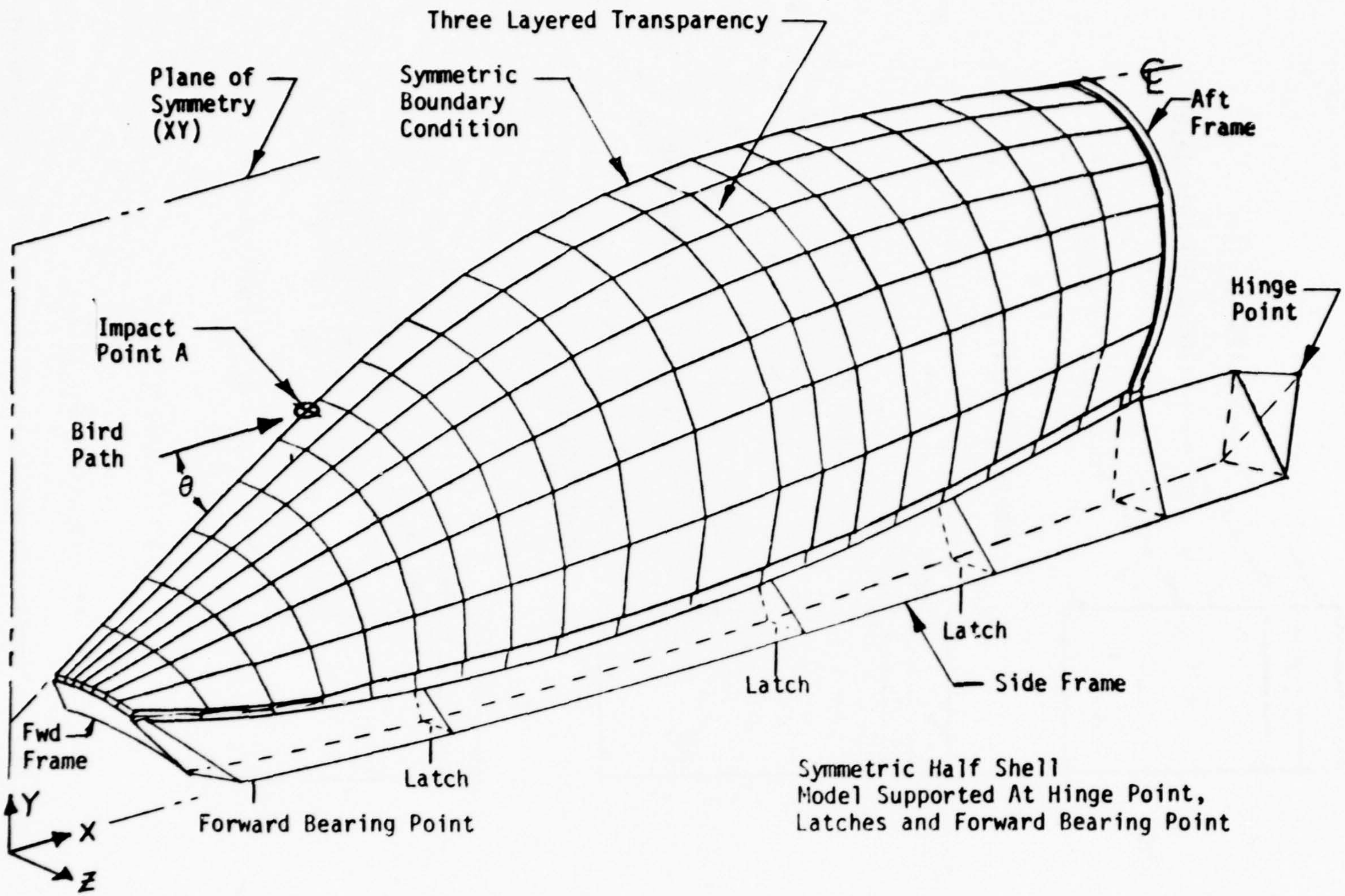
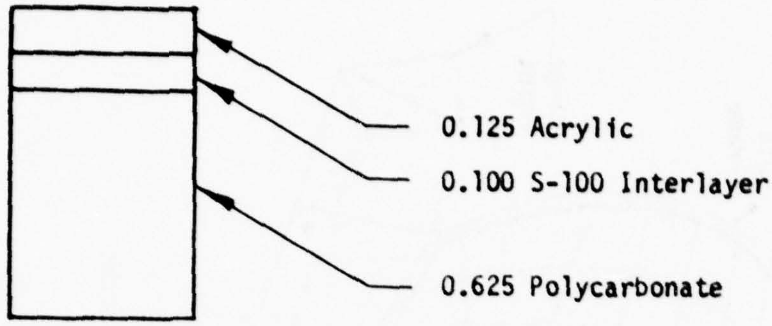
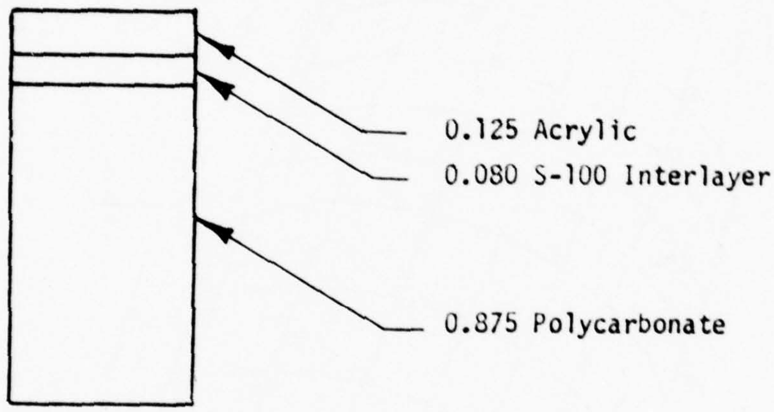


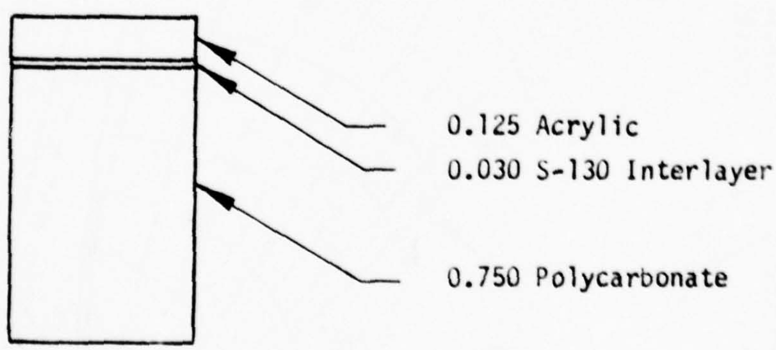
Figure 79. F-16 Finite Element Model, Clear View Configuration.



CONFIGURATION 1



CONFIGURATION 2



CONFIGURATION 3

All dimensions are inches.

Figure 80. Thickness and Materials, F-16 Clear View Canopy Model.

rapidly with the deformation waves. It was not feasible at this stage in the development of the math model to anticipate which elements would be located in the localized high strain rate area at each time increment and provide suitable changes in material properties. Therefore, the best compromise was to use low strain rate properties throughout for the transparency materials. Standard material properties for aluminum, as found in MIL-HDBK-5, were used in the canopy frame elements.

TABLE 23. MATERIAL PROPERTIES(1)

		ACRYLIC	POLYCARBONATE	S-100	S-130
Elastic Modulus	E (PSI)	480,000	270,000	180*	2610*
Secant Modulus,	E_A (PSI)	321,400	143,000	180*	2610*
True Yield Stress,	σ_A (PSI)	9,000	9,076	-	-
True Rupture Stress,	σ_r (PSI)	10,960	11,936	167	1000
True Rupture Strain	ϵ_r(in./in.)	0.0459	0.531	0.928	0.383
Poisson's Ratio,	ν	0.35	0.35	0.45	0.45

(1) Low strain rate properties at room temperature.

* Calculated from: $E = 2(1+\nu)G$. Where G = Modulus of Rigidity.

The bird impact loading was applied to each configuration. The impact was represented by a rectangular force/time distribution, discussed in Reference 27. The average impact force acted on the structure continuously for a specified time. From Reference 27,

$$F_{avg} = \frac{m\ v^2 \sin\theta}{L}$$

$$t_d = \frac{L}{v}$$

where F_{avg} is average impact force

m is bird mass

v is relative velocity of bird to windshield

θ is impact angle

t_d is impact duration time

L is effective length of the bird

It is seen from the equation for F_{avg} and Figure 79 that the average impact force is the component of force normal to the surface of the canopy at the impact point. In these analyses the traveling footprint concept was used by which the impact force moves along the canopy surface with the bird material and the effective length L becomes the distance over which the load footprint is estimated to be in contact with the surface. The loads generator program was used to calculate the distribution of the average impact force to the joints of the model for each of ten time increments and the corresponding positions of the footprint.

For a 4-pound bird traveling at 350 knots,

$$m = \frac{4.0}{386.064} = 0.010361 \ \frac{\text{LB-SEC}^2}{\text{IN.}}$$

$$v = 350\ (20.24) = 7084 \ \frac{\text{IN.}}{\text{SEC}}$$

and for $\theta = 29.23°$ and $L = 25$ inches,

$$F_{avg} = 10{,}156 \text{ LB}$$

$$t_d = 0.003529 \text{ SEC}$$

Since the model is a symmetric half shell, half of the average impact force was applied.

$$1/2\ F_{avg} = 5{,}078 \text{ LB}$$

The first 30 natural free vibration modes were calculated for each of the three configurations and used in transformation matrices to reduce the problem size from 4871 structural degrees of freedom to 30 modal degrees of freedom. Linear incremental response solutions were performed for 23 time increments. The first 10 increments covered the impact duration time of 0.003529 seconds while the remaining ones gave the response out to 0.01 seconds. Modal displacements, velocities, and accelerations and element forces, stresses (including equivalent stresses) and strains for all elements were calculated and stored on magnetic tape for all 23 time points. Postprocess operations transformed the modal displacements to X, Y, Z joint displacements, printed out selected joint displacements along the centerline of symmetry, and printed out stress components, equivalent stresses and strains for selected elements.

Figures 81, 82 and 83 show joint displacements in the Y degree of freedom along the centerline of symmetry at several time points. Note that the average impact force is removed after time point 0.00353 second, and the maximum deflection occurs at or just before the point. Figure 84 shows the strain distribution through the laminate just aft of the impact point. The presence of the thinner, stiffer S-130 interlayer in Configuration 3 causes the laminate to act nearly like a monolithic layer. The maximum deflections and equivalent stresses are tabulated in Table 24. The linear analysis results are corrected by multiplying by a nonlinear correction factor from Figure 85. The data used to develop Figure 85 is tabulated in Table 25, and is based on comparisons between test results and the results of linear math model analyses. The linear analyses were designed to duplicate the tests as nearly as possible. However, the number of complete analyses which could be performed was limited, so that many of the analytical values shown were estimated by use of the equation shown at the bottom of the table.

Note that the corrected results for Configurations 2 and 3 involve the use of Δ_T/Δ_L from Figure 85 for values of $\Delta_L/h < 1$. Points in this region correspond to snap-through. The curve in the snap-through region

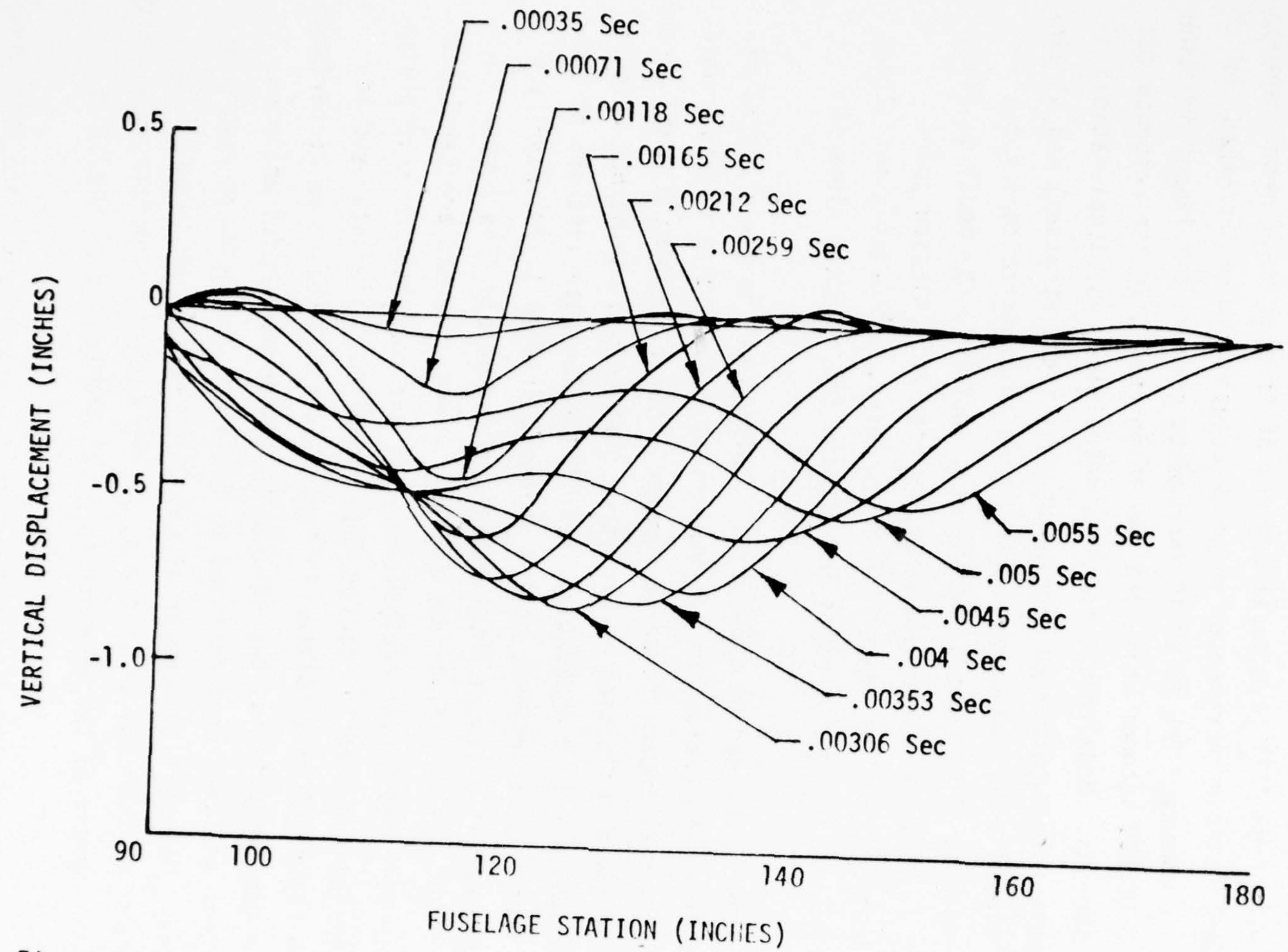


Figure 81. Vertical Displacement on Centerline as a Function of Time for Configuration 1.

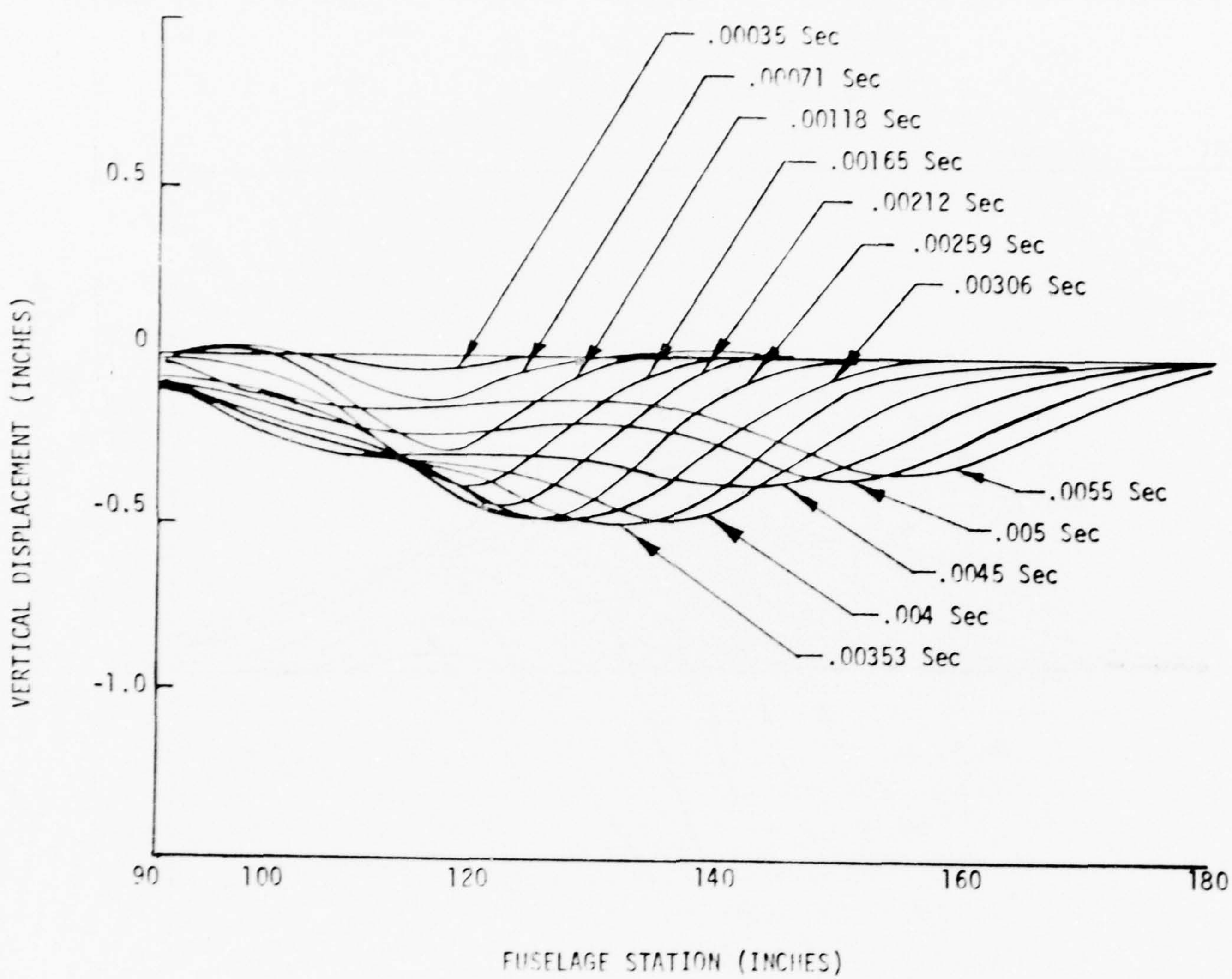


Figure 82. Vertical Displacement on Centerline as a Function of Time for Configuration 2.

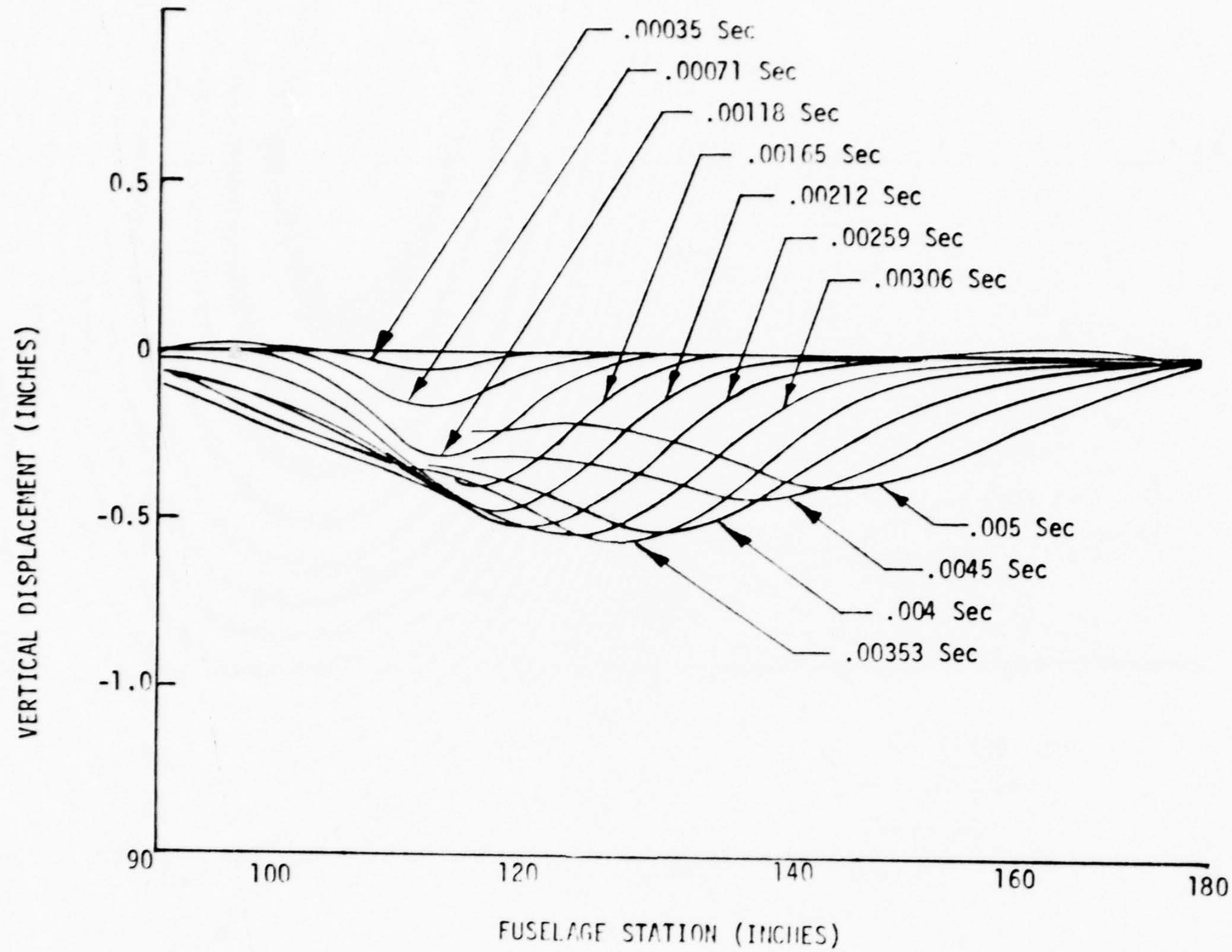


Figure 83. Vertical Displacement on Centerline as a Function of Time for Configuration 3.

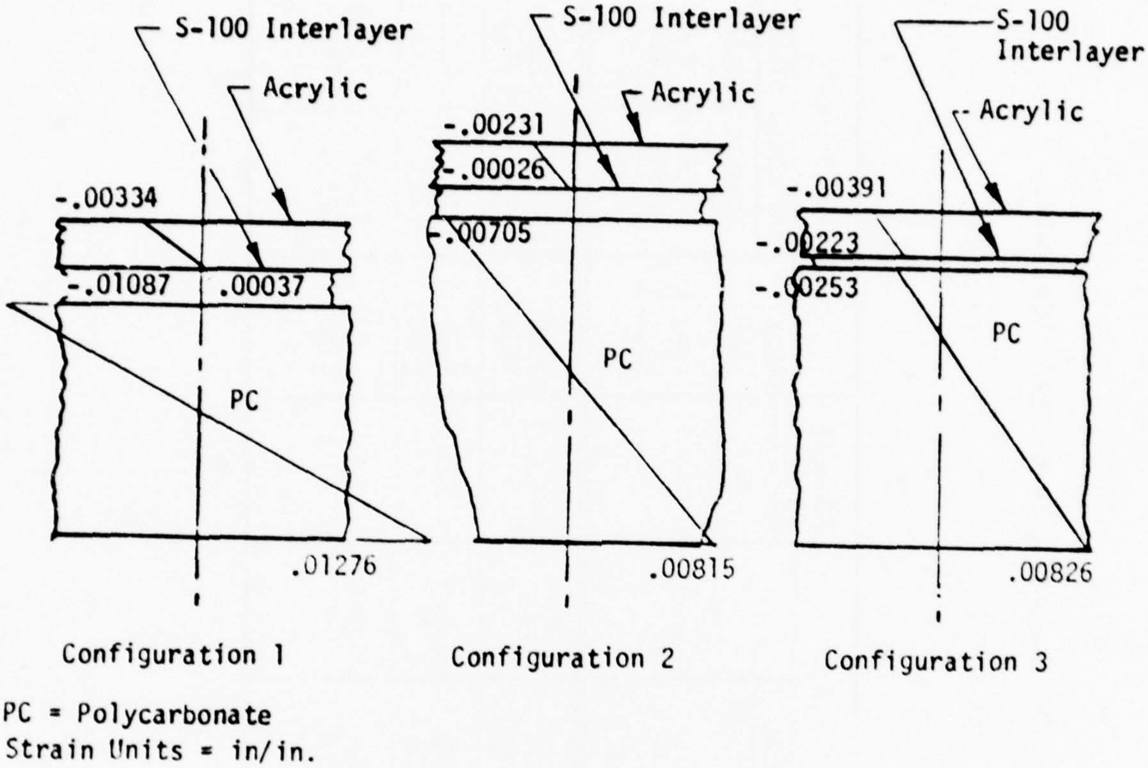


Figure 84. Transverse Strain Distributions, Linear Response Analysis.

is considered less reliable than for larger values of Δ_L/h, in view of the scatter of the test data. Consequently, rather low confidence must be placed in these results.

TABLE 24. SUMMARY OF MATH MODEL ANALYSIS RESULTS, F-16 CLEAR VIEW CANOPY

CONFIG	LINEAR ANALYSIS RESULTS					NON-LIN CORRECT	CORRECTED RESULTS		
	POLY-CARB. h (IN.)	MAX LIN DEFL Δ_L (IN.) *[F.S.]	MAX EQUIV STRESS ACRYLIC $\sigma_{L_{ACR}}$ (PSI) [F.S.]	MAX EQUIV STRESS POLYCARB $\sigma_{L_{P.C.}}$ (PSI) [F.S.]	$\frac{\Delta_L}{h}$	** $\frac{\Delta_T}{\Delta_L}$	MAX DEFL Δ (IN.) [F.S.]	MAX EQUIV STRESS ACRYLIC $\bar{\sigma}_{ACR}$ (PSI) [F.S.]	MAX EQUIV STRESS POLYCARB $\bar{\sigma}_{P.C.}$ (PSI) [F.S.]
1	0.625	0.82 [124]	4815 [119]	3905 [116]	1.31	4.5	3.69 [124]	21,670 [119]	17,570 [116]
2	0.875	0.50 [130]	2878 [119]	2730 [119]	0.57	2.5	1.25 [130]	7,195 [119]	6,825 [119]
3	0.750	0.57 [130]	4528 [119]	2406 [116]	0.76	3.2	1.82 [130]	14,490 [119]	7,700 [116]

* [F.S.] = Fuselage Station

** Reference Figure 85.

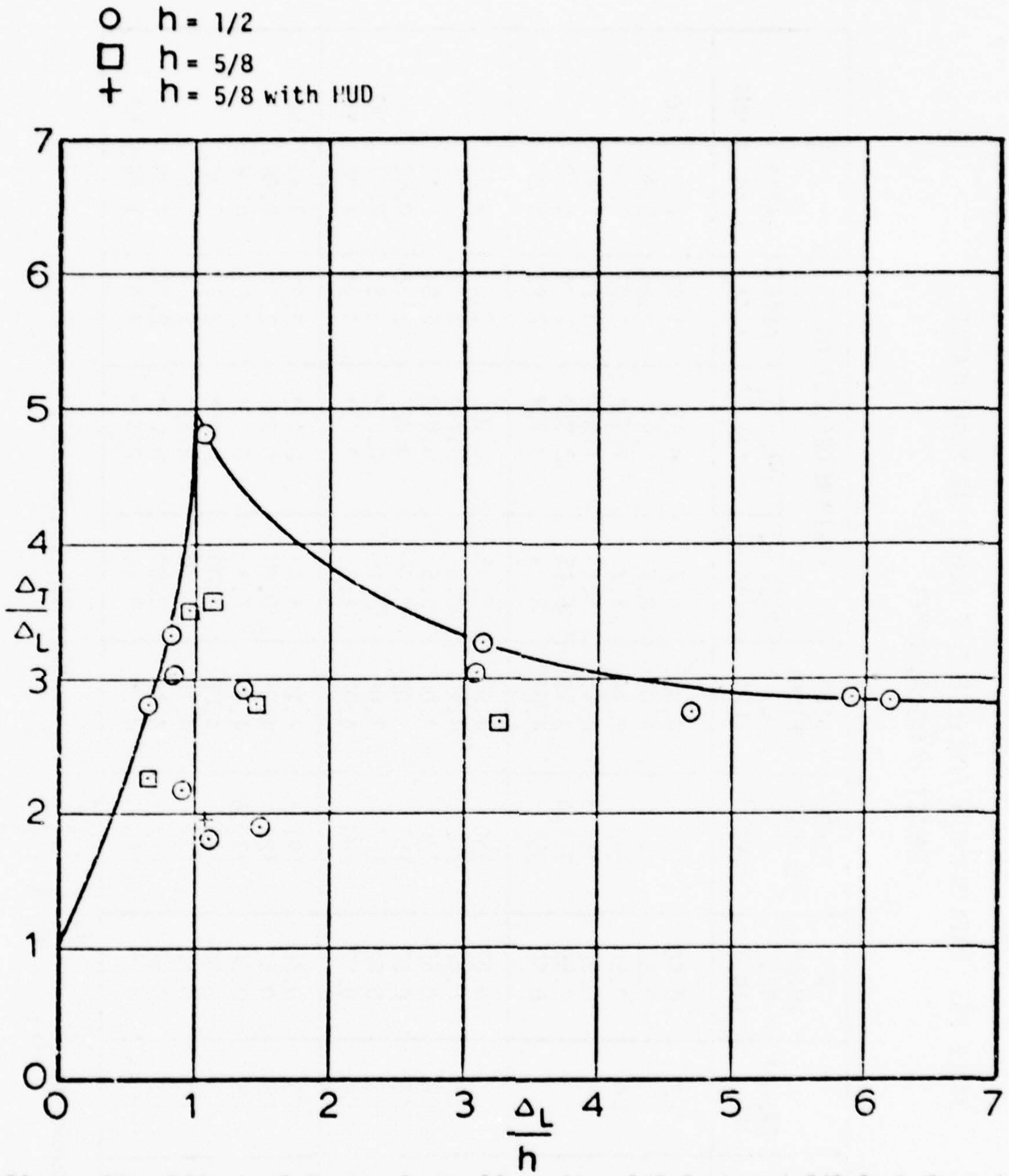


Figure 85. Effect of Geometric Nonlinearity, 1/2-Inch and 5/3-Inch Canopies.

TABLE 25. DATA SUMMARY FOR EFFECT OF GEOMETRIC NONLINEARITY

(IMPACT POINT A, 75°F)

LINEAR ANALYSIS RUN NO.	TEST NO.	CANOPY NO.	h THICKNESS (IN.)	BIRD VELOCITY v (KT)	BIRD WEIGHT w (LB)	MAXIMUM DEFLECTION TEST Δ_T (IN.)	MAXIMUM DEFLECTION ANAL. Δ_L (IN.)	MAXIMUM DEFLECTION RATIO Δ_T/Δ_L	RATIO Δ_L/h	HUD
26	011	C2	0.50	176	4.07	1.4	0.75	1.87	1.5	NO
28	GD3		0.50	340	4.02	8.4	2.94	2.86	5.88	
	006	C4	0.50	342	2.16	4.7	1.55*	3.03	3.10	
	GD1		0.50	344.3	2.16	5.1	1.57*	3.25	3.14	
	GD2		0.50	351.4	3.12	6.45	2.35*	2.74	4.70	
	GD5		0.50	355	4.00	8.75	3.10*	2.82	6.20	
25	022F		0.62	350	4.00	5.35	2.03	2.64	3.27	
	022	C8	0.62	202	4.10	2.5	0.70*	3.57	1.13	
	022D	C8	0.62	230	4.00	2.5	0.89*	2.81	1.44	
27	028	C6	0.62	157	4.02	0.95	0.42	2.26	0.68	
	023	C6	0.62	187	4.05	2.1	0.60*	3.50	0.97	NO
	022C	C8	0.62	200	4.00	1.3	0.67*	1.94	1.08	YES
	012	C5	0.50	191	2.10	1.4	0.42*	3.33	0.84	NO
	013	C5	0.50	123	4.06	0.95	0.34*	2.79	0.68	
	013A	C5	0.50	139.6	4.01	1.3	0.43*	3.02	0.86	
	018	C3	0.50	146	4.00	1.00	0.46*	2.17	0.92	
	020A	C3	0.50	157	4.18	1.00	0.56*	1.79	1.12	
	020B	C3	0.50	173	4.18	2.00	0.69*	2.90	1.38	
	GD4	0056	0.50	156	4.00	2.55	0.53*	4.81	1.06	NO

*Estimated Based on $\frac{\Delta_{L2}}{\Delta_{L1}} = \frac{w_2}{w_1}\left(\frac{v_2}{v_1}\right)^2$ Using Runs 26, 28, 25, 27.

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SECTION XI
TRANSPARENCY DESIGN CONCEPTS
ADAPTABLE TO THE F-16 CANOPY DESIGN

This section presents the design concepts that have been defined as potential candidates for the F-16A and F-16B canopies to meet various bird impact test levels of 350 through 562 knots, and also a design that is applicable to the Air Force Manufacturing Technology Program (MANTECH).

The basic cross section specified by contract for this program was a three-ply laminate consisting of an acrylic outer ply and a polycarbonate structural ply joined by an interlayer. The thickness of the polycarbonate is varied to meet the various bird impact levels of 350, 400, 430, 450, 500 and 562 knots for the high visibility type of canopy design.

Other canopy design concepts, including utilization of an integral bow frame and a fixed windshield concept, satisfy the bird impact requirements at these higher velocities. Two canopy design concepts which are premised on an altered canopy shape to meet the 350-knot bird impact requirement and various other criteria, are presented for consideration. In each case where the transparency thickness is increased, the inner surface is retained and the outer surface is moved outboard.

Initially, it was anticipated that the bird impact math model would be utilized as a design tool, after appropriate correlation with test data, to analyze the potential F-16 canopy changes needed to design a bird impact resistant transparency for impact levels of 350 through 562 knots. The attempted nonlinear analyses of the F-16 canopy were unsuccessful. Reasons for the failure are discussed in Reference 26.

BACKGROUND

This effort was initiated at the request of the Aeronautical Systems Division when it became apparent that the coating used to protect the polycarbonate canopy in the YF-16 prototype aircraft eroded from the

exterior canopy surface after relative short exposures to rain and ice crystal abrasion. Similar experience on the F-111 and F-15 canopies substantiated the necessity of further development to provide improvements in protecting polycarbonate from the environment. Recognition of this problem resulted in the F-16 Alternate Canopy Program. The objectives of this program were: (1) to generate and experimentally validate a transparent canopy design incorporating environmental durability significantly superior to coated monolithic polycarbonate, (2) to test the initial production F-16A canopy configuration and to assess the potential for pilot injury due to a four-pound birdstrike at flight speeds up to 350 knots, and (3) to support a program that was established to develop manufacturing methods for forming an F-16A laminated canopy.

The configuration selected to fulfill the first (1) objective consisted of an acrylic outer ply, an interlayer, and a polycarbonate inner ply. This design was required to be adaptable to the F-16 aircraft as well as future high performance aircraft and was required to be capable of defeating a four-pound birdstrike at 350 knots. Additional design refinements were required for birdstrike protection at flight speeds up to 562 knots.

In order to accomplish the second (2) objective, a series of bird impact tests were conducted on the initial production F-16A canopy to determine bird impact location sensitivity, establish impact resistance to smaller birds, and ascertain the effect of canopy thickness. The results of these tests are reported in Section VII.

A series of studies were generated to validate specific aspects of the proposed concept for the F-16 Alternate Canopy Program. The ensuing tests and test results are presented in a set of reports which are summarized in Section IV of this report.

The results of thermal studies that were conducted in support of the proposed design are presented in Section V of this report. Section VI

describes material studies that were performed in support of the selected materials. Cyclic edge joint tests were conducted to verify the structural integrity of the attachment area for a laminated design and are reported in Section VIII. The effect of ice crystal impact on face ply material is reported in Section IX of this report. Section X describes the application of a math model/computer program to only a linear analysis for laminated concepts. The final math model, it is believed, must account for nonlinear conditions to be successful.

CONCEPT FOR 350-KNOT BIRD IMPACT

The basic cross section selected to provide environmental durability superior to coated monolithic polycarbonate was a three-ply laminate consisting of an acrylic outer ply and a polycarbonate structural ply joined by a silicone interlayer. The acrylic ply thickness is 0.080 inch, the interlayer thickness is 0.080 inch, and the minimum thickness required for the polycarbonate is 0.740 inch. This configuration is shown in Figure 86. It is recommended that the formed polycarbonate thickness should not be less than 82 percent of the preformed material stock thickness in order to meet the bird impact capability. The recommended maximum polycarbonate thickness is 0.810 inch.

Design Features

The initial production F-16 transparency was a 0.500-inch thick monolithic polycarbonate with the canopy Mold Line (ML) on the outboard surface. The proposed thicker laminated transparency will have the added thickness go outboard of the ML and the inner surface will move outboard by 0.07 inch. This configuration is shown in Figure 87. The trim of the interlayer and acrylic is recessed from the edge of the polycarbonate. This concept avoids attachments through the acrylic and interlayer. A retainer of corrosion resistance (cres) steel is used to seal the edges and to aid in preventing delamination. See Figure 88.

The canopy side fairing was redesigned as shown in Figure 89. Weather sealing is provided by a bulb seal bonded to the fairing. To restrain the

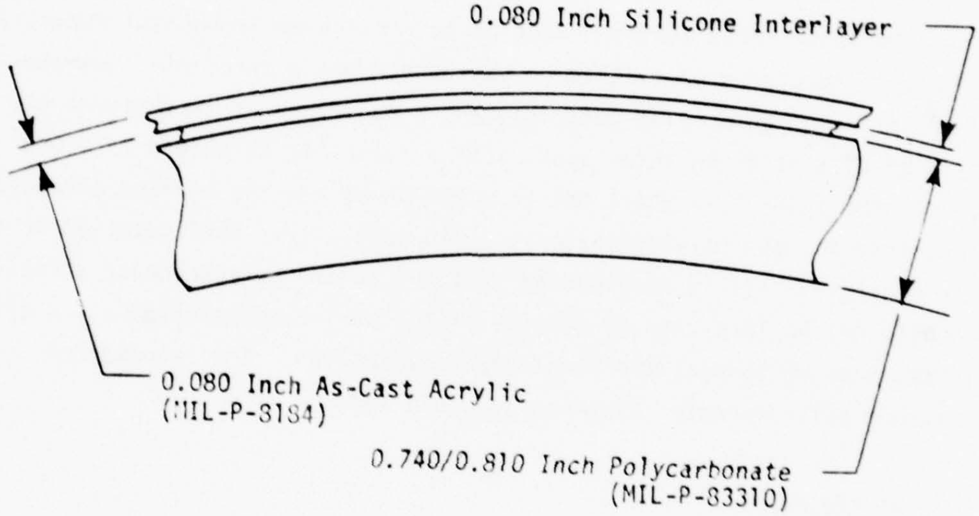


Figure 86. Laminate Cross Section.

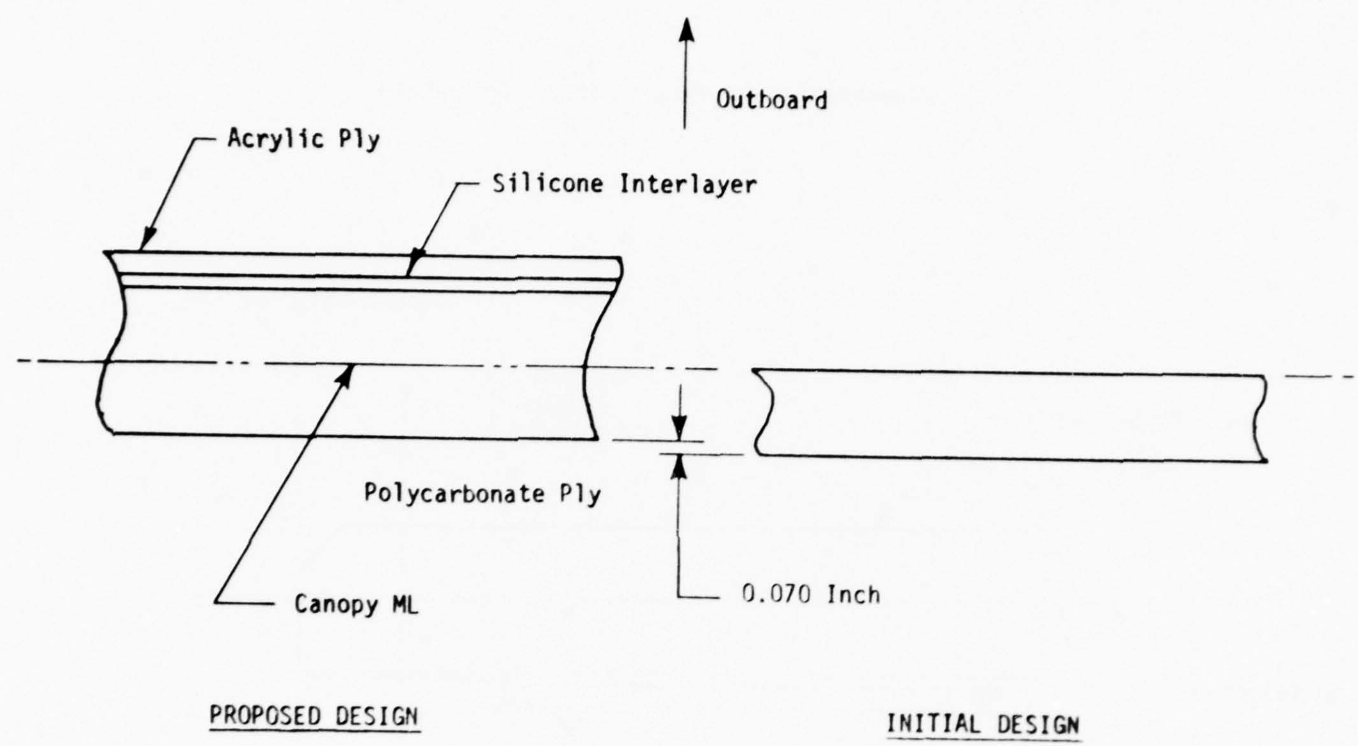


Figure 87. Added Thickness Outboard of Canopy Mold Line (ML).

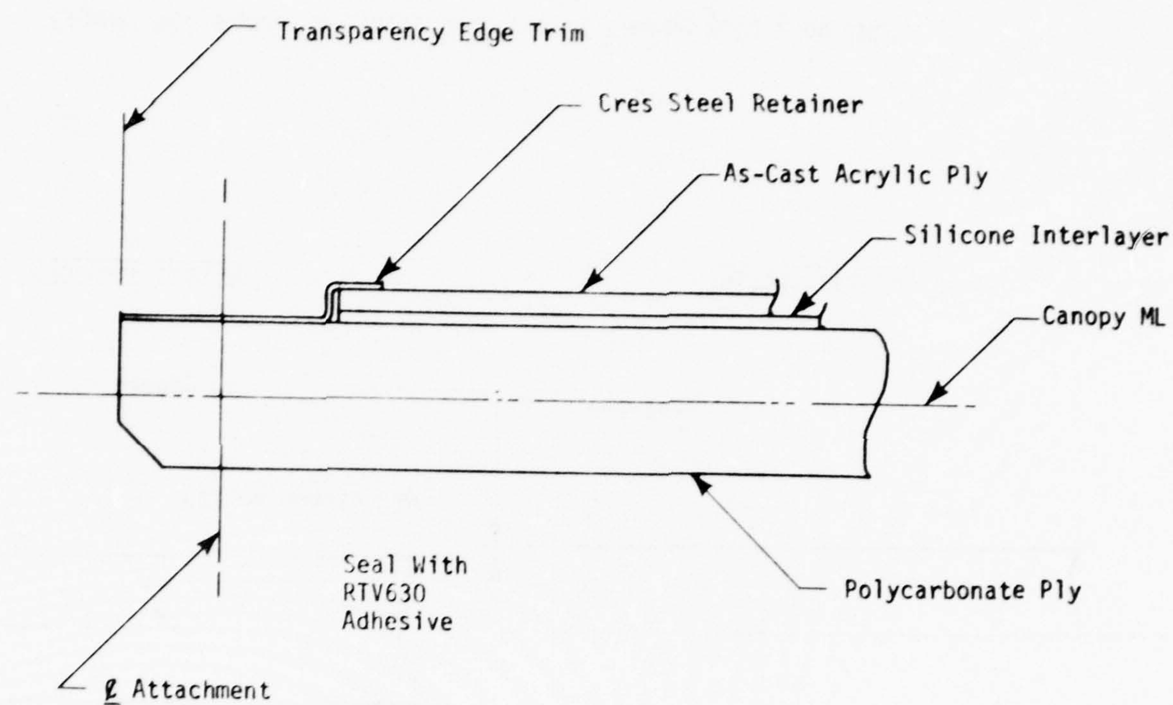


Figure 88. Typical Edge of Transparency.

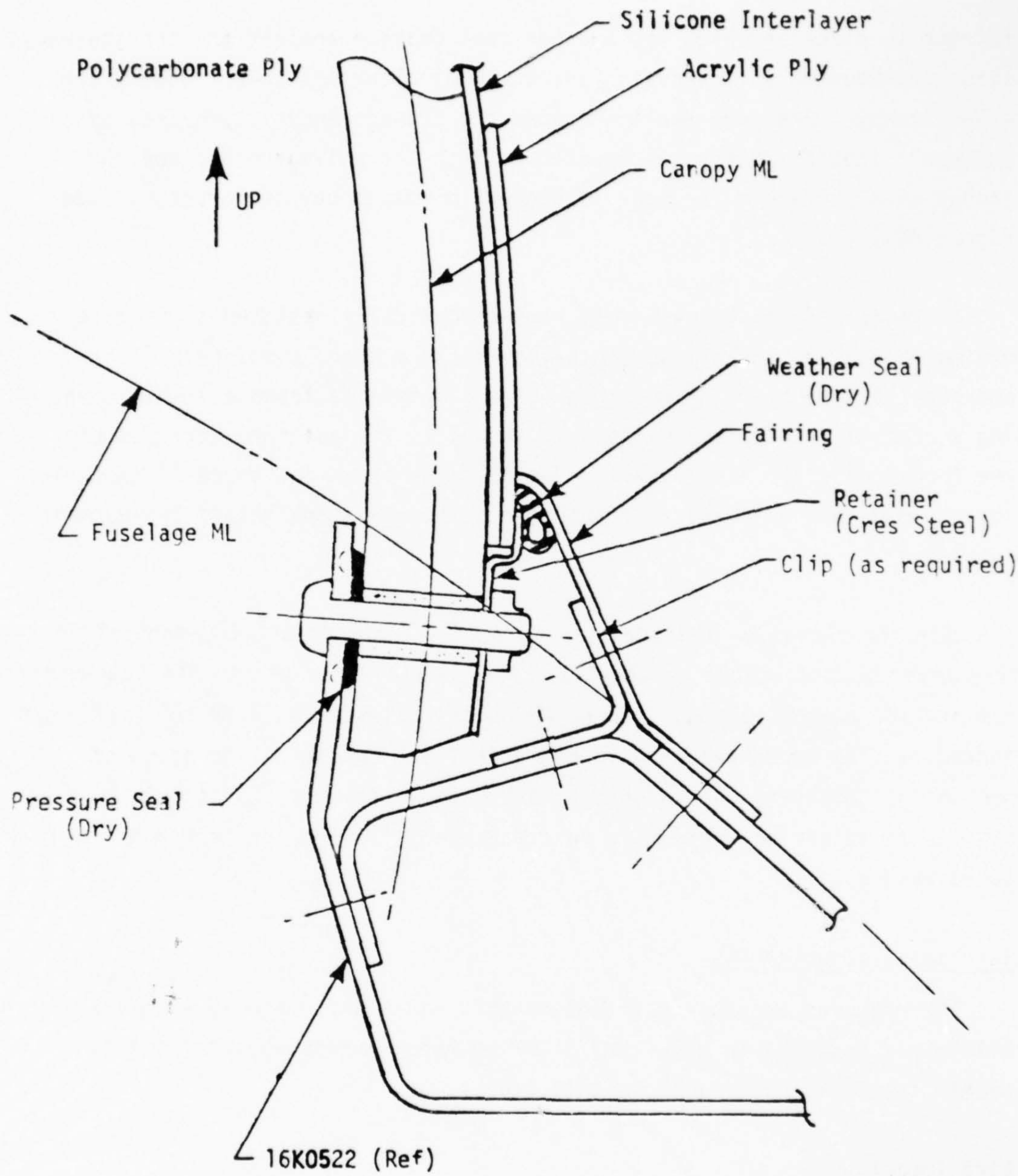


Figure 89. Typical Section Along Lower Edge of Canopy.

fairing in place and keep the weather seal bearing against the transparency, clips are mounted at intervals (as required) along the lower edge of the transparency. Pressure sealing around the transparency is provided by a dumb-bell shaped rubber seal mounted between the polycarbonate and the canopy substructure. The seal is bonded to the canopy substructure (see Figure 89).

The edges of the transparency were conceptually designed to produce minimal effect on the production canopy substructure to maintain interchangeability as close as possible. The canopy side frame assembly mounting surface is virtually unaffected. However, the outboard surfaces of the forward and aft frames were redesigned, as shown in Figures 90 and 91, to accommodate the increased transparency thickness and retain aerodynamic fairing.

With the increased thickness (Figure 86), based on a 0.500 monolithic transparency, the weight of the transparency will increase to 154 pounds minimum and 166 pounds maximum (depending on the tolerance), a 78 to 92 percent increase. The added weight will not effect the canopy's kinematics of mechanism. However, the canopy actuator and jettison rocket should be studied to determine whether or not its capabilities are affected by the added weight.

Structural Requirements

The proposed concept described in this section for the 350-knot capability was designed to meet the following requirements when fabricated within the limits described in this report.

Bird Impact

The transparency, when mated to an appropriate canopy structure and fuselage support structure, is capable of withstanding, without penetration or causing pilot incapacitation, the impact of a four-pound bird at a velocity of Mach 0.53 (350 knots). The point of impact is defined as being on the centerline and 14 inches forward of the eye level/centerline point as measured along the outer surface of the transparency. The transparency

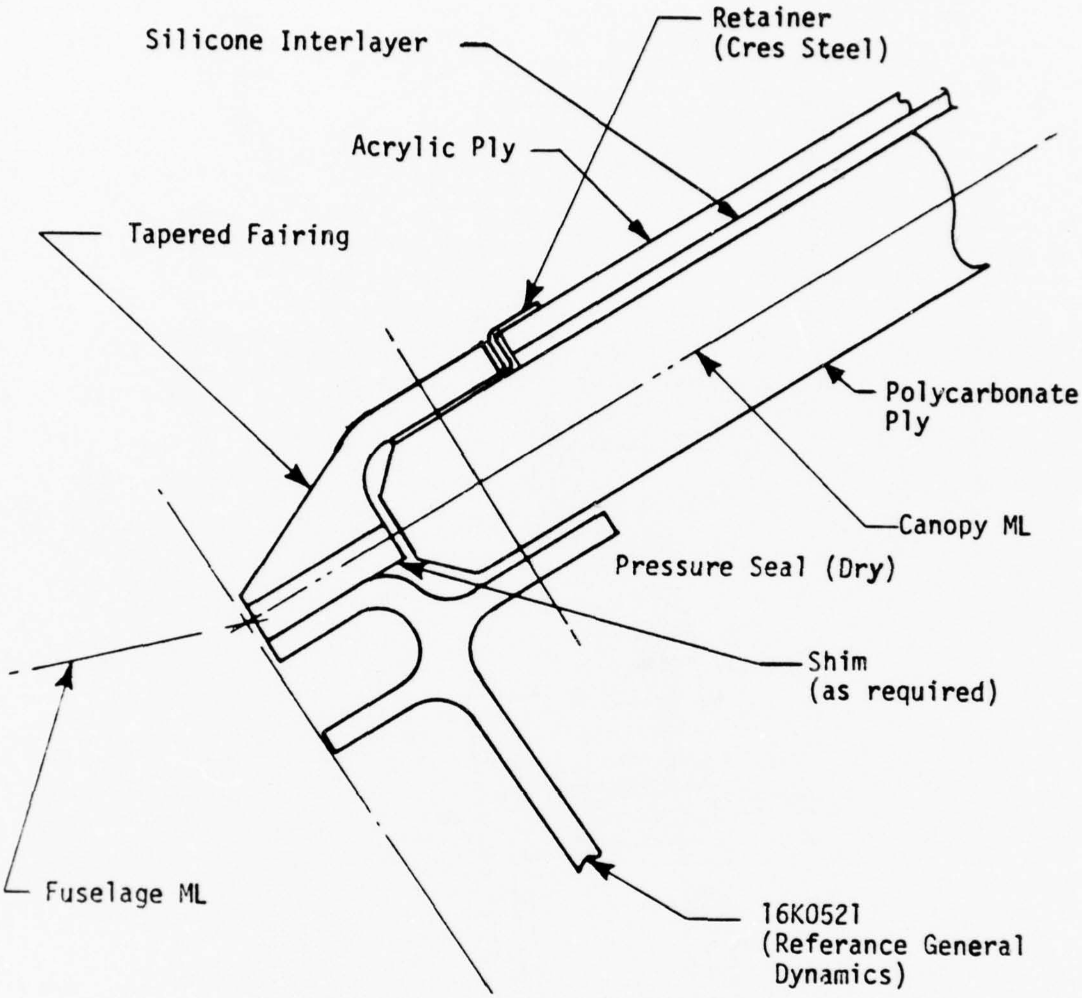


Figure 90. Fairing Forward Edge of Transparency.

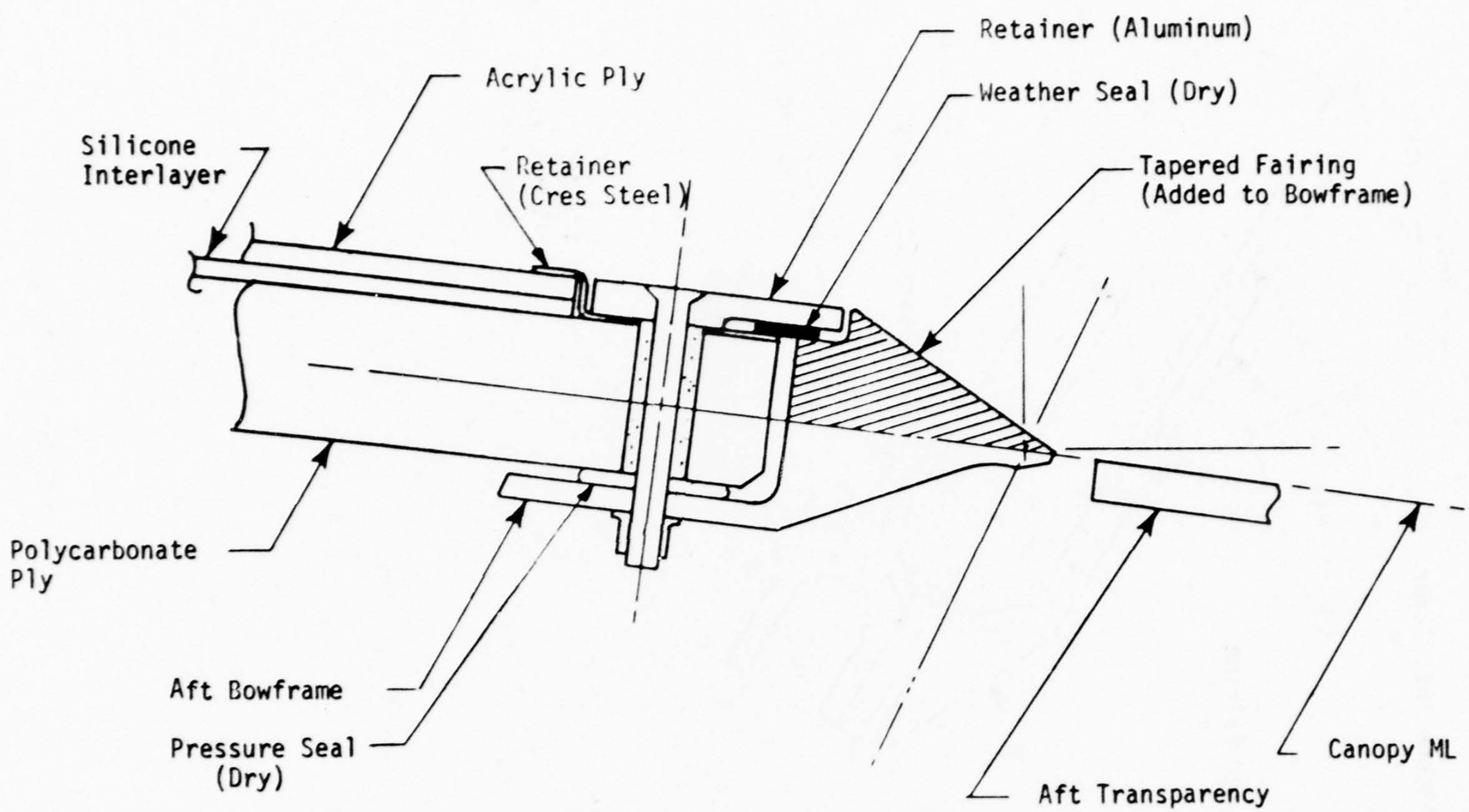


Figure 91. Fairing Aft Edge of Transparency.

will meet this requirement with an average temperature of -35°F or 195°F throughout the transparency.

Structural Integrity

The transparency, when mated to an appropriate canopy structure and fuselage support structure, will sustain the static requirements noted below and the accelerated environmental test of four lifetimes of the cyclic pressure and associated temperatures specified in Section VIII of this report without structural damage. One lifetime shall consist of 1814 cycles which corresponds to 2000 flight hours.

Proof Pressure - The transparency will sustain 6.8 psig internal air pressure at room temperature for a period of fifteen (15) minutes without damage, delamination or edge separation.

Operational Ultimate Pressure - The transparency will sustain a maximum operational pressure of 12.5 psi. The operational ultimate pressure of 12.5 psi is the sum of the maximum design internal pressure, the external pressure, and the port reference error pressure. The pressure can be held for 6 minutes at 2/3 maximum pressure and at maximum pressure (12.5 psi) for 30 seconds at an average temperature of 200°F throughout the transparency.

Edge Attachment - The transparency edge and attachment will sustain an ultimate load in single shear tension of 350 pounds per attachment at a temperature of 200°F along the longitudinal edges.

Structural Analysis

The following structural analysis addresses the durability of the face ply material, the capability of the interlayer to couple the face ply to the structural ply and its ability to survive the extreme thermal conditions, and the structural integrity of the three ply configuration to sustain a bird impact.

An analysis of the proposed concept to meet the proof pressure and ultimate pressure requirements is not included. This concept is considered to be stronger than the previous 1/2 inch canopy since the structural ply is thicker.

The edge joint analysis is presented in Section VIII of this report.

Face Ply

The outer ply for this proposed cross section is as-cast acrylic (MIL-P-8184), 0.080-inch thick. According to MIL-P-8184 the minimum thickness of as-cast acrylic in sheets large enough to meet the size requirements of the F-16 transparency is 0.125 inch. Two approaches are recommended to acquire large sheets of 0.080-inch acrylic: canvas the plastic manufacturers for premium sheet availability or consider machining from 0.125-inch thickness to 0.080-inch thickness.

If the 0.125 inch thick acrylic is used as the laminate face ply, it would be possible to balance the increase in acrylic thickness with a decrease in the polycarbonate thickness of approximately 0.030-inch since the stiffness of the cross section is a function of both thicknesses and the coupling capability of the interlayer. A weight penalty would be incurred since the polycarbonate works harder than the acrylic. The net weight increase would be approximately 3 pounds.

Acrylic of 0.080 inch thickness was tested during this program in a wind tunnel as noted in Reference 25 and under impact as noted in Reference 28, to verify the materials reliability under adverse conditions. Results from these tests supported the hypothesis that 0.125 inch thickness is not essential from a material characteristics standpoint.

Section IX of this report presents the results of a salt abrader test conducted in support of the selected face ply material.

It is considered that hail impact protection would be adequate for the proposed design as noted in Section II of this report.

Interlayer

The interlayer chosen for this laminated cross section, 0.080 inch thick, is cast-in-place silicone (Sierracin S-100). The silicone interlayer couples the outer ply to the inner polycarbonate ply and increases the effective stiffness of the total cross section. The coupling effectiveness of the interlayer decreases as the interlayer temperature increases as noted in Technical Report, AFFDL-TR-77-96 (Reference 18). The 0.080 inch thickness is considered to be minimum thickness acceptable for the design requirements as explained in the following paragraph.

Section V of this report presents a thermal analysis for two laminated configurations, one with 0.080-inch silicone interlayer and the other with 0.100-inch silicone interlayer. The analysis was based on a canopy length of 90 inches and a constant temperature from front to back. The differential thermal expansion at the maximum temperature will elongate the acrylic ply more than the polycarbonate ply and will result in an offset of 0.128 inch for the 0.080 inch interlayer and a shear strain of 1.6 inches/inch. Data from Reference 18 indicates that the rupture point for silicone interlayer at 195°F, low strain rate, occurs at an average strain of 1.6 inches/inch. The thermal analysis indicates a 1.36 inches/inch strain for the 0.100-inch silicone interlayer. Although a reduction in the thickness of the interlayer does increase the interlayer coupling action and increases the effective beam thickness, the 0.080 inch interlayer was chosen as the minimum acceptable interlayer thickness since the thermal analysis indicates that this design is marginally acceptable.

A companion design, using a 0.030-inch copolymer interlayer (Sierracin S-130), was investigated for potential use in a laminated design. This material was discarded as a candidate based on the following reasons:

- Preliminary data from Technical Report AFFDL-TR-77-96 (Reference 18) indicates that the S-130 copolymer will fail adhesively under bird impact loading.

- Thermal studies included in Section V of this report indicate that stresses exist beyond the interlayer rupture point at high temperatures due to differential elongation of the acrylic ply and polycarbonate ply.

- Edge joint tests reported under Section VIII of this report resulted in bubbling of the S-130 copolymer interlayer.

Structural Ply

The inner ply is the principle load carrying member of the laminated transparency design and is polycarbonate sheet (MIL-P-83310). The bird impact test results described in Section VII of this report established 0.760 inch (Figure 56) as the thickness of monolithic polycarbonate required to sustain the impact of a four-pound bird at Location A (eye level, centerline) for 350 knots, 75°F temperature, and will limit the eye point transparency deflection to less than two inches. This value was increased to 0.780 inch to accept an impact at Location B for monolithic polycarbonate and represents a weight increase of approximately four pounds. Location B is approximately 14 inches closer to the forward edge of the transparency than Location A. The minimum thickness of polycarbonate required for the laminated design is 0.740-inch as shown in Figure 86, for impact at Location B.

The decrease in the required polycarbonate thickness for the laminated design (0.740 inch versus 0.780 inch for monolithic) is made possible by the coupling of the acrylic face ply to the polycarbonate ply by the silicone interlayer mentioned previously. The effective stiffness of the laminated section was calculated by application of the method described in AFFDL-TR-76-114, "The Determination of Deflection and Stress Distribution for a Transparent laminated beam"(Reference 29). This computer program

was written for flat beams and the analysis indicates that the beam effective stiffness increases as the beam length increases. Therefore, the effective beam length was determined by considering the deflection plot, Figure 92, for a 4.02-pound bird impact at 363 knots, 75°F, on a 0.50-inch polycarbonate transparency (Test 022F, Section VII). The transparency had deflected 2.0 inches. The point of inflection for the deflected shape was 10.5 inches from the centerline and a length of 21 inches was chosen as the beam length for calculating the effective beam stiffness.

An iterative analyses was conducted using the computer program of AFFDL-TR-76-114 (Reference 29) to construct a laminated cross section that would have an equivalent thickness equal to 0.780-inch, since 0.780 is considered to be the thickness of polycarbonate required to defeat a four-pound bird at 350 knots. A cross section consisting of 0.080-inch acrylic face ply, 0.080-inch silicone interlayer, and a 0.740-inch polycarbonate structural ply was found to have the required equivalent thickness of 0.780. The effective stiffness for this section was 15,280 lb-in.^{2} for a beam length of 21 inches and a width of 1.00 inch. This value was converted to an equivalent beam thickness as follows:

$$EI = 15{,}820 \text{ lb-in.}^2$$

$$EI = \dot{E}\,\frac{t_{eq}^3}{12} \qquad \text{where } \dot{E} = 400{,}000 \text{ PSI}$$

$$\therefore \; t_{eq} = 0.780\text{-inch}$$

The value of 400,000 PSI for E was determined from test data in Reference 18 and is based on high strain rate testing at room temperature (72°F). The value used for the shear modulus of the silicone interlayer was 146 PSI and is based on a high strain rate at 72 °F. This value was also based on test data in Reference 18.

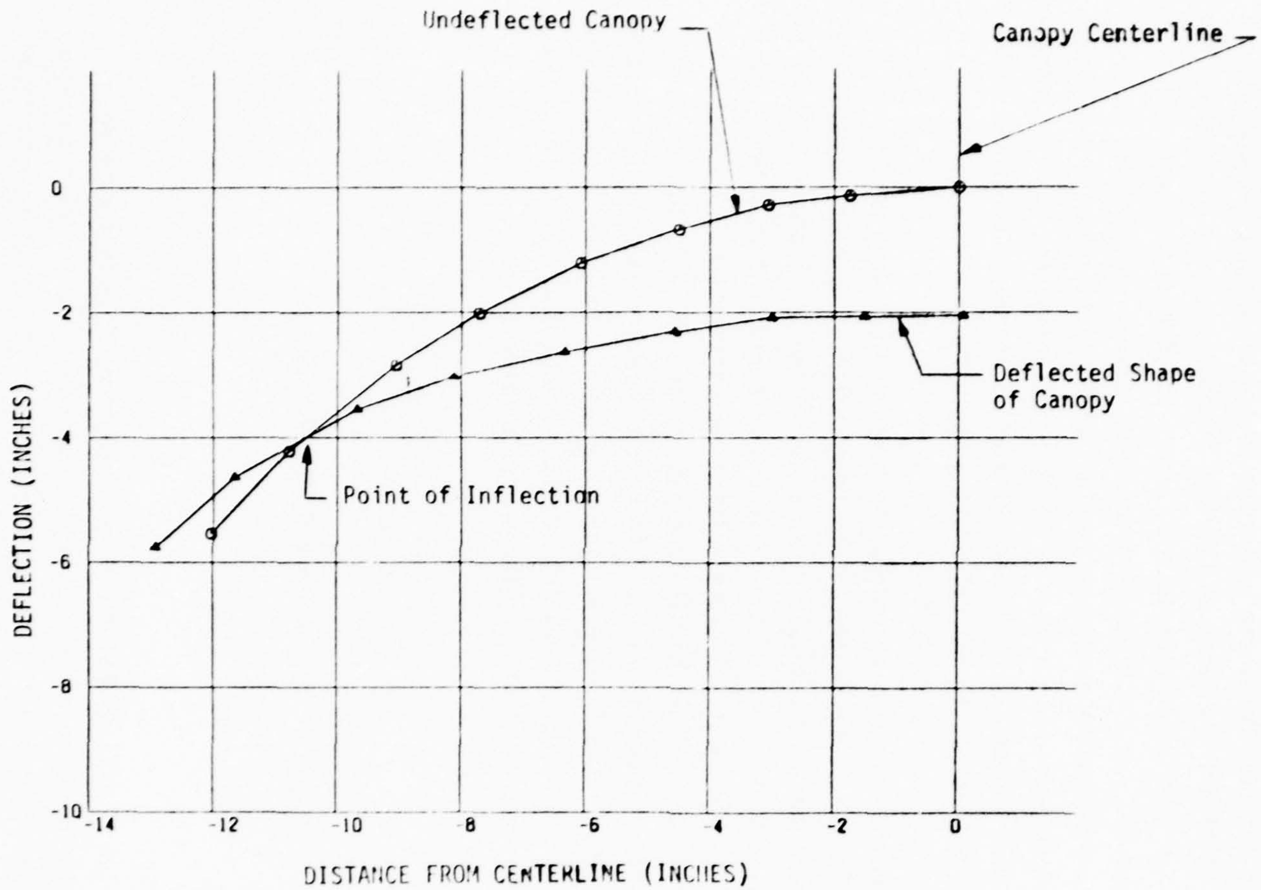


Figure 92. Transparency Deflection, Station 137.50 (Test 022G, Section VII).

Vision/Optical Analysis

Vision

The proposed concept, being thicker than the initial production F-16 monolithic transparency, results in an increase in depth to the forward and aft fairings, and results in a change to the geometry of the side fairings. At a zero azimuth the down vision will decrease by less than 1 percent, and at the 90 degree azimuth the down vision will decrease by less than 3 percent. These deviations are believed to be acceptable for the proposed design.

Optical

The calculated percent light transmission, defined in Section II for a configuration similar to the proposed concept, is 70.45 percent for the installed position of 30 degrees (with zero surface curvature). This is approximately 2 percent less than the calculated 1/2 inch polycarbonate configuration. This value may be compared to test data, reported in Reference 25, which records a light transmission value of over 87 percent for 1/2 inch coated polycarbonate when measured normal to the surface.

Section II of this report defines a minimum acceptable light transmission value as 65 percent at the installed angle. It is believed that the proposed concept would meet this requirement.

Haze requirements are defined in Section II as 4 percent maximum when measured normal to the surface. Values are shown in Reference 25 which vary from 1.26 percent to 3.28 percent (the average is 2.4 percent) for 1/2 inch and 5/8 inch coated polycarbonate. Values are shown for laminated configurations which vary from 2.12 percent to 6.48 percent (4.9 percent average). These laminated configurations consist of 0.08 inch as-cast acrylic, 0.10 inch silicone interlayer, and 1/2 inch and 5/8 inch polycarbonate. It is believed that the proposed concept would meet the haze requirements of 4 percent when fabricated to strict quality control methods.

Further studies would be appropriate to investigate the deviation and distortion characteristics of the proposed design.

Thermal Analysis

The proposed concept meets the requirements of MIL-E-38453A which limits the inboard transparency surface temperature to a maximum of 160°F.

The acrylic face ply and interlayer reduce the temperature of the structural ply to a level below that of a monolithic polycarbonate configuration, thus reducing the susceptibility of the polycarbonate to thermal embrittlement.

The results of additional thermal studies are reported in Section V of this report.

Electromagnetic Environment Design Considerations

The tactical use of the aircraft has simplified the extent of the electromagnetic environments that must be considered. These are now reduced to precipitation static electricity and lightning.

The proposed canopy design contains no conductive coating or non-structural surface conductors. The thickness of the composite plies is adequate to prevent static electric or lightning puncture. Static electric charging will take place on the outer frontal area. The static discharge path is to the metallic frame members surrounding the open area of the transparency.

If lightning were to strike the canopy area the ionized flash path is expected to be over the outer surface. The lack of conductive coating on or in the transparency will reduce the tendency of the flash to hug

the surface of the canopy. The peripheral metal frame of the canopy should be capable of handling the discharge. However, the concerns regarding the electromagnetic and mechanical forces the flash may have on the canopy-fuselage interface still exist as do the concerns for the effects of the induced magnetic fields. These were described in Section II and must be addressed on an aircraft system basis, as they cannot be solved by the canopy design alone. The electric shock potential due to the retained static electric charge on the non-conductive transparency material must be addressed in handling and operations procedures of the aircraft.

Maintainability

The proposed design improves the overall system maintainability through the use of dry seals, thereby eliminating the long periods of curing time associated with the use of wet sealant.

Further improvements would be realized by reducing the total number of fasteners required for transparency installation. Appropriate tests would be required to structurally prove the integrity of a system with increased bolt spacing.

DESIGN FOR MANUFACTURING TECHNOLOGY PROGRAM (MANTECH)

Drawing Number Z5943254-503 provides the necessary instructions and specifications to fabricate and install a single piece laminated transparency applicable to the Manufacturing Technology Program (MANTECH). This transparency, Figure 93, can be installed on the current F-16A canopy substructure provided the side fairing along the lower edge is omitted. The side fairing is not required for bird impact testing. Standard hardware may be used for the mechanical attachments for canopy installation. To assure proper pressurization of the cockpit area for environmental testing, RTV630 sealant should be applied around the entire periphery of the transparency-canopy substructure joint area as shown in Figure 94.

Differences between the conceptual design and the MANTECH design (Z5943254-503) are as follows:

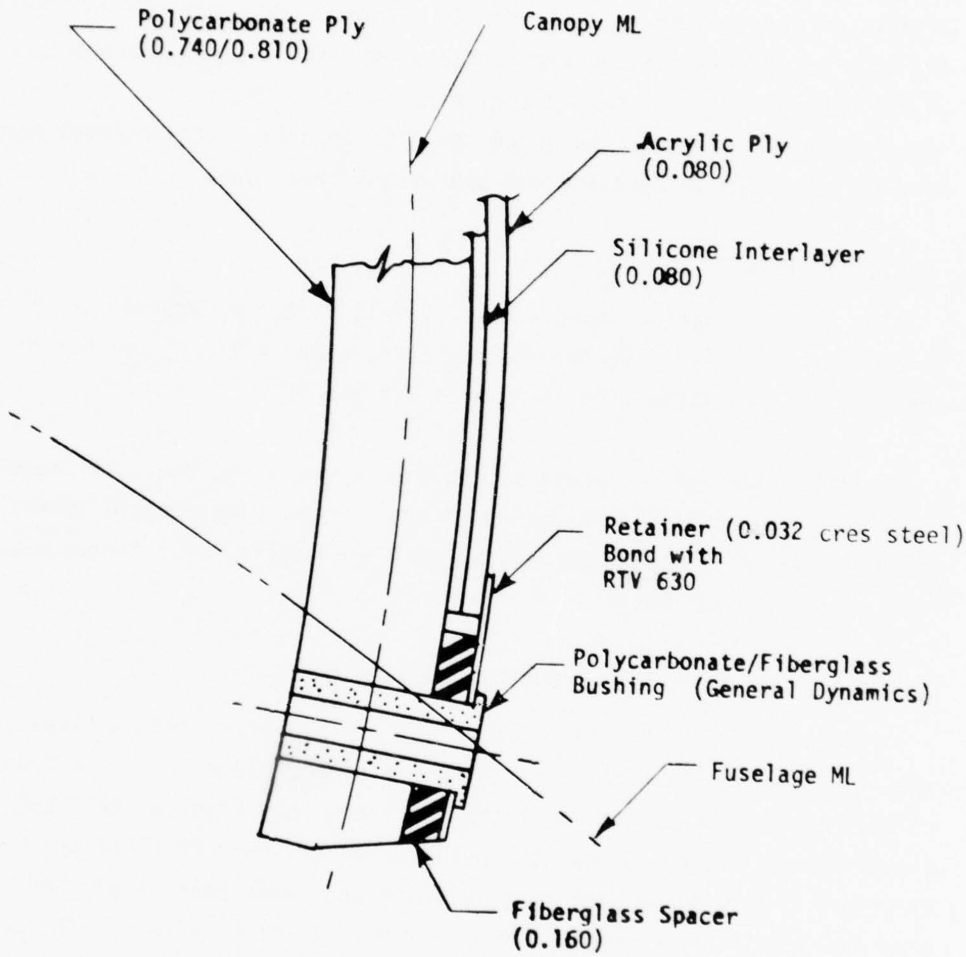


Figure 93. Concept for Manufacturing Technology Program.

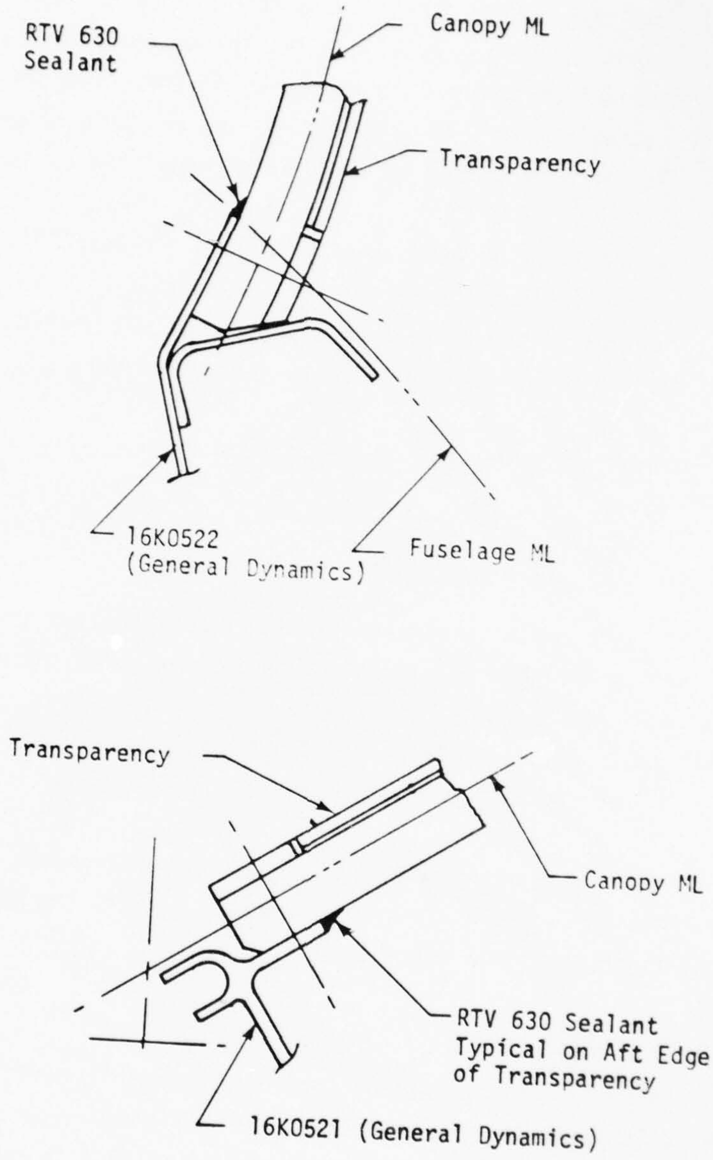


Figure 94. Applying RTV 630 Sealant.

a) The edge design shown on Z5943254-503 has a nylon strip spacer along the periphery of the transparency with a flat retainer sealing the spacer and the edges of the laminates. In the conceptual design, the nylon strip spacer has been omitted and the retainer is formed to seal the edges of the laminate (Figure 88).

b) Wet sealant is used when installing the Z5943254-503 transparency to seal the cockpit area for pressurization. In the conceptual design, a dry seal is used in lieu of wet sealant to eliminate the curing time required for wet sealant (Figure 89).

This design is capable of defeating a four-pound bird at a velocity of 350 knots.

CONCEPTS FOR BIRD IMPACT LEVELS ABOVE 350 KNOTS

The basic cross section specified by contract for this program was a three-ply laminate consisting of an acrylic outer ply and a polycarbonate structural ply joined by an interlayer. The thickness of the polycarbonate is varied to meet the various bird impact levels of 400, 430, 450, 500 and 562 knots for the current high visibility type of canopy design.

Other canopy design concepts, including utilization of an integral bow frame and a fixed windshield concept, satisfy the bird impact requirements at these higher velocities.

High Visibility Design

The cross section defined for this concept is 0.080 inch acrylic outer ply, 0.80 inch silicone interlayer and a polycarbonate structural ply. The thickness of the polycarbonate ply is defined in Table 26 for the various impact velocities. These monolithic polycarbonate thicknesses were derived by extrapolating from the bird impact test results, Section VII of this report. The rationale for the selection of the acrylic thickness, the interlayer thickness, and the stated reduced polycarbonate thickness for the laminated configuration, is given in the preceding paragraphs that

TABLE 26. POLYCARBONATE THICKNESS REQUIREMENTS FOR DESIGN VELOCITIES (1)

VELOCITY (2) (KNOTS)	THICKNESS (3) +0.070,-0 (INCHES) MONOLITHIC	THICKNESS (4) +0.070,-0 (INCHES) LAMINATED CONFIGURATION	WEIGHT FOR LAMINATED CONFIGURATION (POUNDS) (5)
350	0.78	0.74	154/166
400	0.85	0.81	166/178
430	0.89	0.85	173/185
450	0.91	0.87	177/189
500	0.97	0.93	187/199
562	1.03	0.99	197/210

(1) The formed polycarbonate shall not be less than 82 percent of the preformed thickness as specified.

(2) Impact at Location B, 4-pound bird, 75°F maximum 2.0 inch deflection at eye point. Location B is 14 inches forward of Location A (eye level centerline) measured on transparency surface.

(3) Derived from bird impact test results.

(4) The laminated configuration consists of an acrylic face ply (0.080 inch), a silicone interlayer (0.080 inch), and a polycarbonate structural ply.

(5) The weight is based on a constant thickness.

describe the concept for the 350-knot level. The data given in Table 26 is for a four-pound bird, impact Point B, 75°F temperature, and a maximum eye point deflection of 2.0 inches. The formed polycarbonate should not be less than 82 percent of the preformed thickness as specified. The minimum and maximum weights are shown in pounds and are based on a constant thickness of transparency. Location B is 14 inches forward of Location A (eye level, centerline) measured on the transparency surface.

Integral Bow Frame

Two concepts are proposed for this design: a one-piece transparency with an aluminum frame bonded to the inner canopy surface, and a two-piece transparency joined by an aluminum frame.

One Piece Transparency

This concept would utilize the same canopy cross section as depicted in Figure 86, but the polycarbonate thickness could be decreased. This concept is shown in Figure 95 with the epoxy nomex filler strip bonded to the inner surface of the canopy and the fixed bow frame. This integral stiffener, located just forward of the pilot, will reduce the deflection of the canopy and prevent contact with the pilot's head.

The maximum bird penetration velocity for this concept has not been established. The ratio of the transparency stiffness to the bow frame stiffness is critical. It is expected that this design would provide sufficient strength to meet the birdstrike requirements at higher energy levels with thinner structural plies than those required for the high visibility design.

The advantage of this concept is less overall weight. The disadvantage of this concept is less vision when compared to the high visibility design.

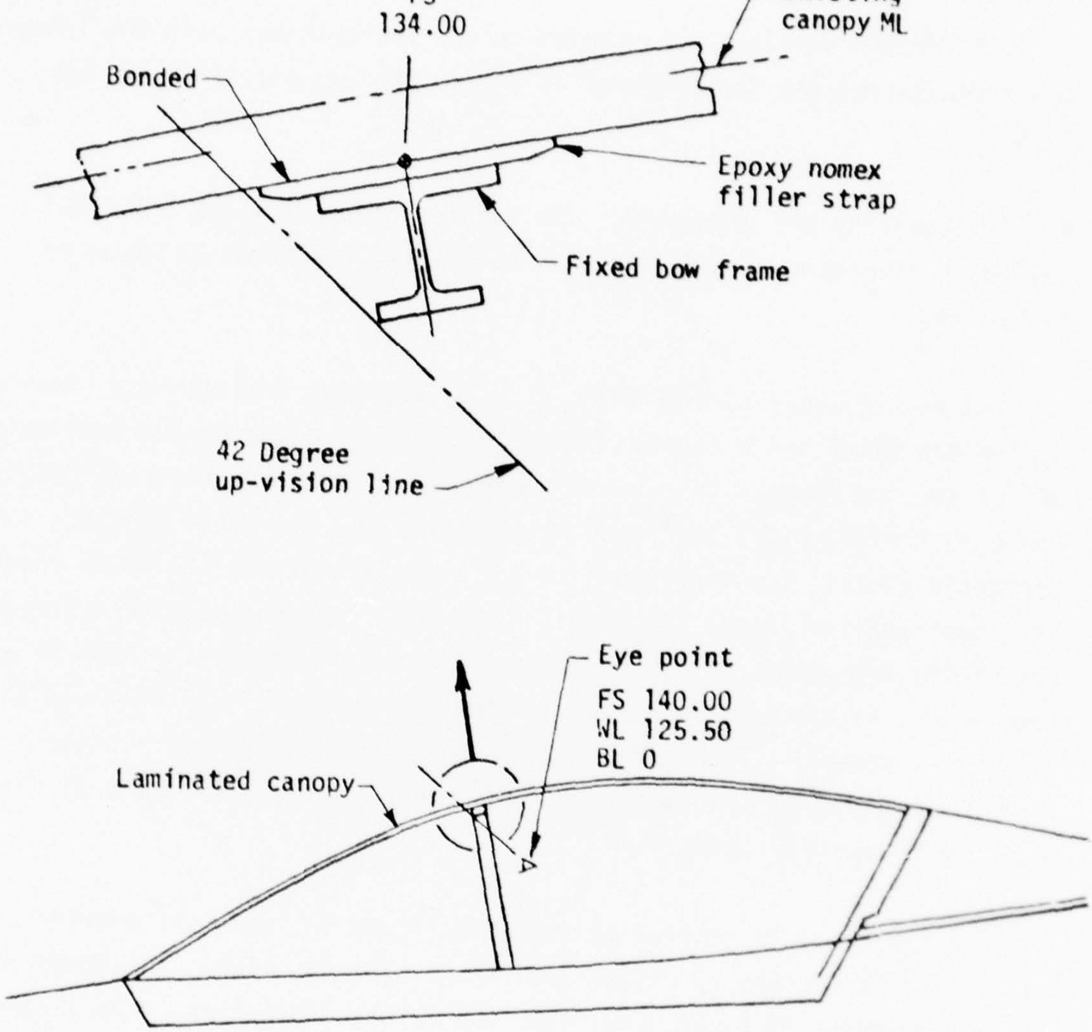


Figure 95. Laminated Canopy With Integral Bow Frame Design Concept

Two Piece Transparency

This design is similar to the one piece transparency with the integral bow frame except the transparency is split into two pieces at the bow frame.

Two concepts are presented. The first concept utilizes identical laminated transparencies forward and aft of the bow frame as shown in Figure 96.

The second concept, Figure 97, uses a laminated transparency forward of the bow frame and a thinner coated monolithic polycarbonate transparency aft of the bow frame. This concept weighs less than the high visibility concept, the one piece bow frame concept, and the two piece concept presented above. The small bird-impact angle aft of the bow frame results in a much smaller impact force which permits a reduced material thickness aft of the bow frame. It is considered that abrasion will be less of a problem aft of the bow frame, and the use of coated polycarbonate in this area is acceptable. This concept will produce an optical displacement of the image due to the change in cross section at the bow frame as viewed from the design eye point.

The bow frame is located as shown in Figure 98, and will provide a minimum of 15 degrees up vision from the design eyepoint. The lower edge of the bow frame is located to clear the canopy latch area. The vision envelope complies with the requirements of MIL-STD-850B.

The advantages of the two piece bow frame concept are reduced weight, improved maintainability, and improved manufacturing capability. This concept is expected to provide sufficient strength to meet birdstrike requirements at higher energy levels while requiring less transparency materials. It is required to balance the ratio of transparency stiffness to the stiffness of the bow frame. Maintainability is improved because only the forward transparency requires a laminated configuration.

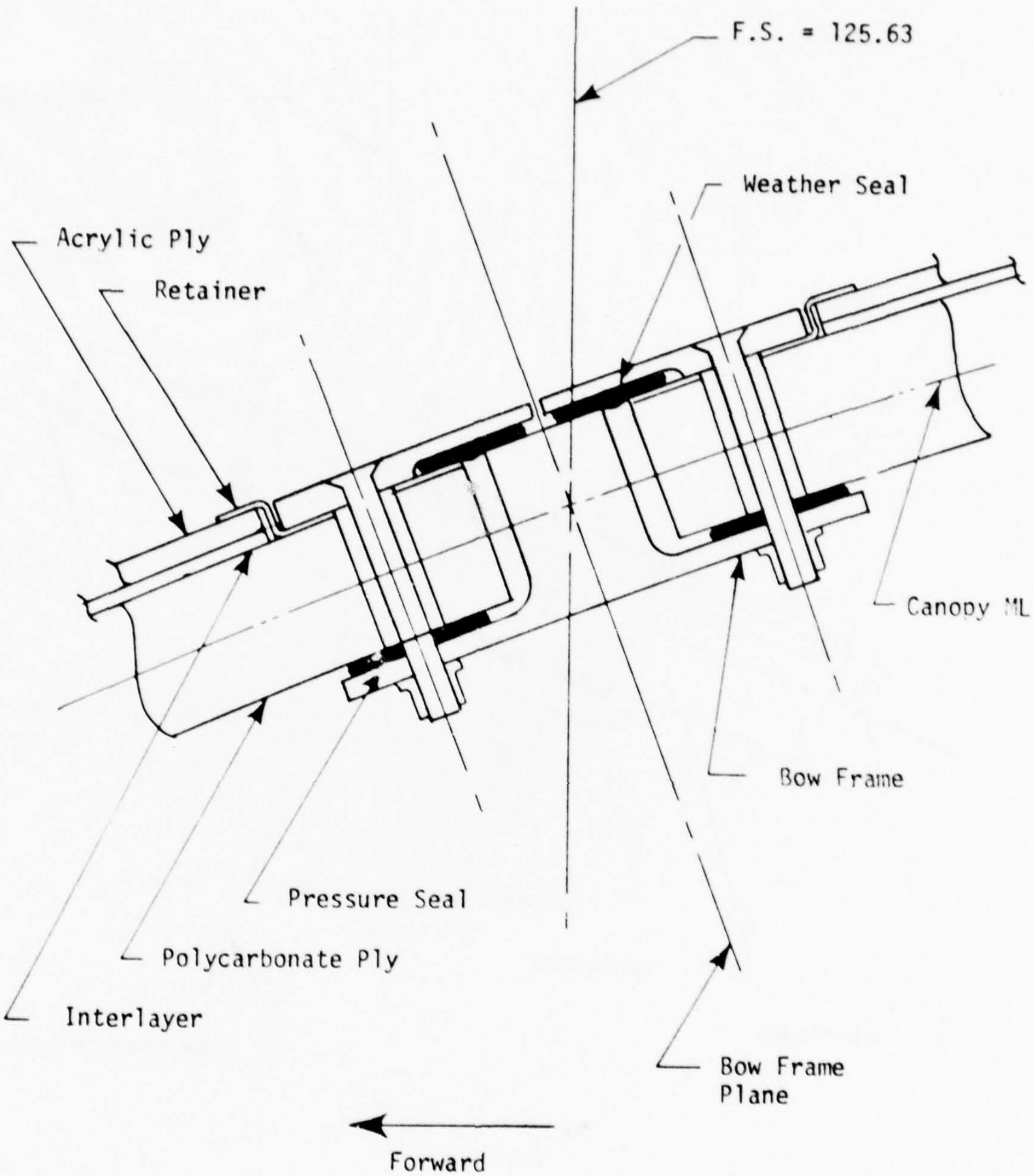


Figure 96. Two Piece Laminated Transparency with Integral Bow Frame.

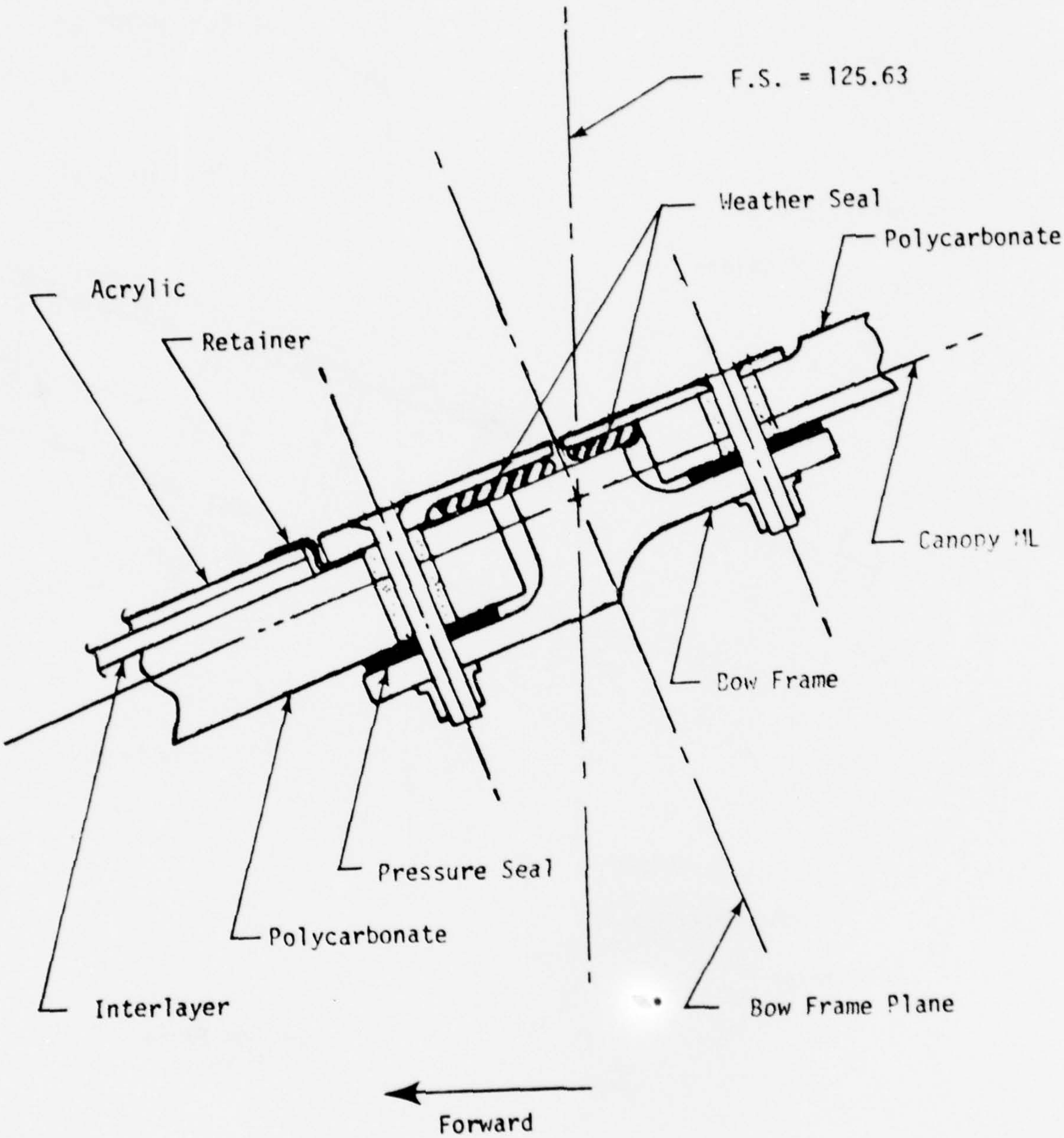


Figure 97. Two Piece Laminated/Monolithic Transparency with Integral Bow Frame.

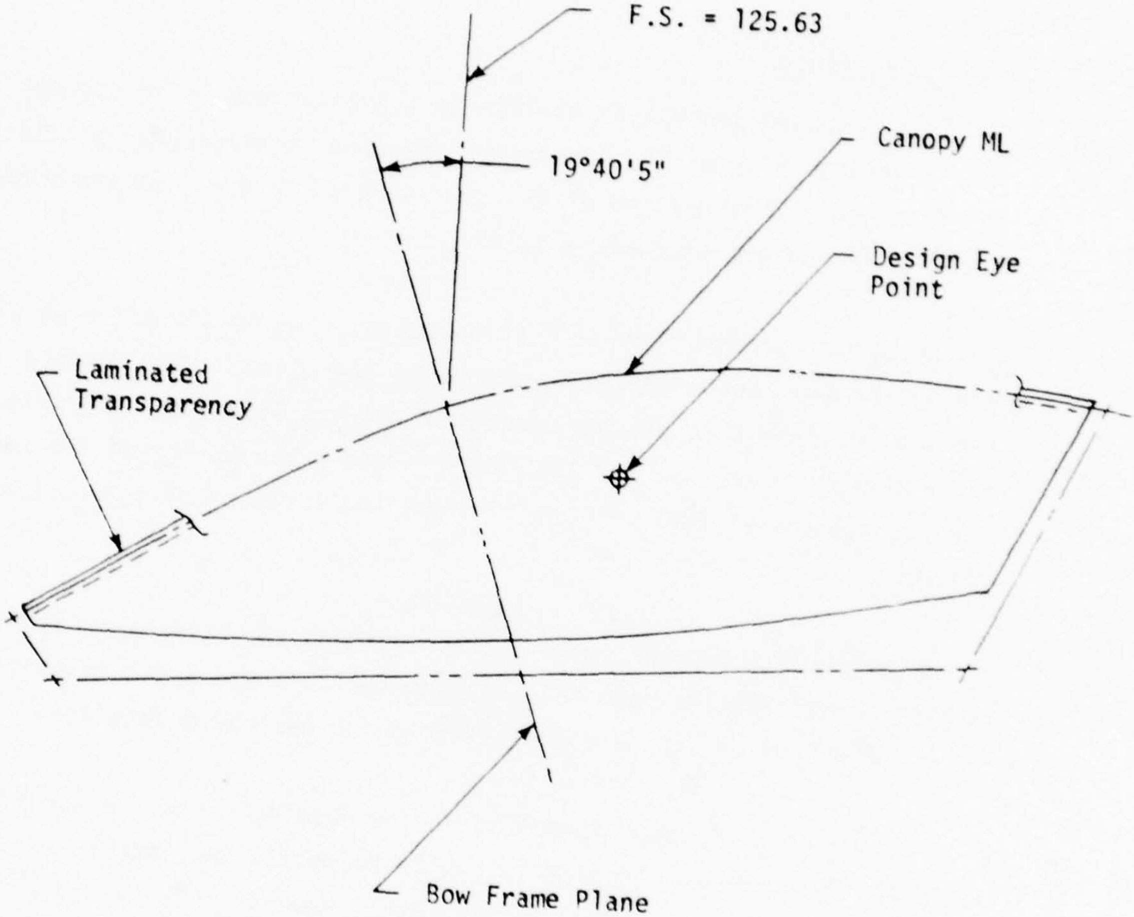


Figure 98. Bow Frame Location.

The disadvantages of this concept include reduced pilot vision and additional parts to fabricate.

Fixed Windshield

This proposed concept is similar to the fixed bow frame concept except the forward portion of the canopy/transparency is structurally attached to the fuselage. This portion of the transparency is known as the windshield. The area aft of the windshield is the canopy.

There are two bow frames in this concept, one on the aft edge of the windshield and one on the forward edge of the canopy. The surface between the two bow frames is the parting plane and its location is dictated by: (1) emergency ejection, (2) the geometry and hinge points of the canopy, and (3) pilot's vision. Figure 99 shows the location of the parting plane.

Figure 100 shows the cross section between the windshield and canopy transparency with the rain and pressure seal. These pressure seals, as in the 350-knot design, are dry seals to improve maintainability.

The windshield, when installed on the fuselage, must be structurally adequate to provide sufficient strength to resist the imposed loads of pressure, bird strike, aerodynamics, etc. However, the windshield must also be readily removable for access to the pilot's console for repairs and maintenance. Figure 101 shows a method that requires the removal of the fairing and windshield fasteners. An alternate method would be to hinge the windshield on the forward edge for quick access to cockpit equipment and latch the sides.

To adapt the fixed windshield concept to the current F-16 will require extensive redesign to the current canopy and fuselage substructure. The areas of potential redesign can be divided into three main groups: Structure, Mechanism, and Canopy Jettison

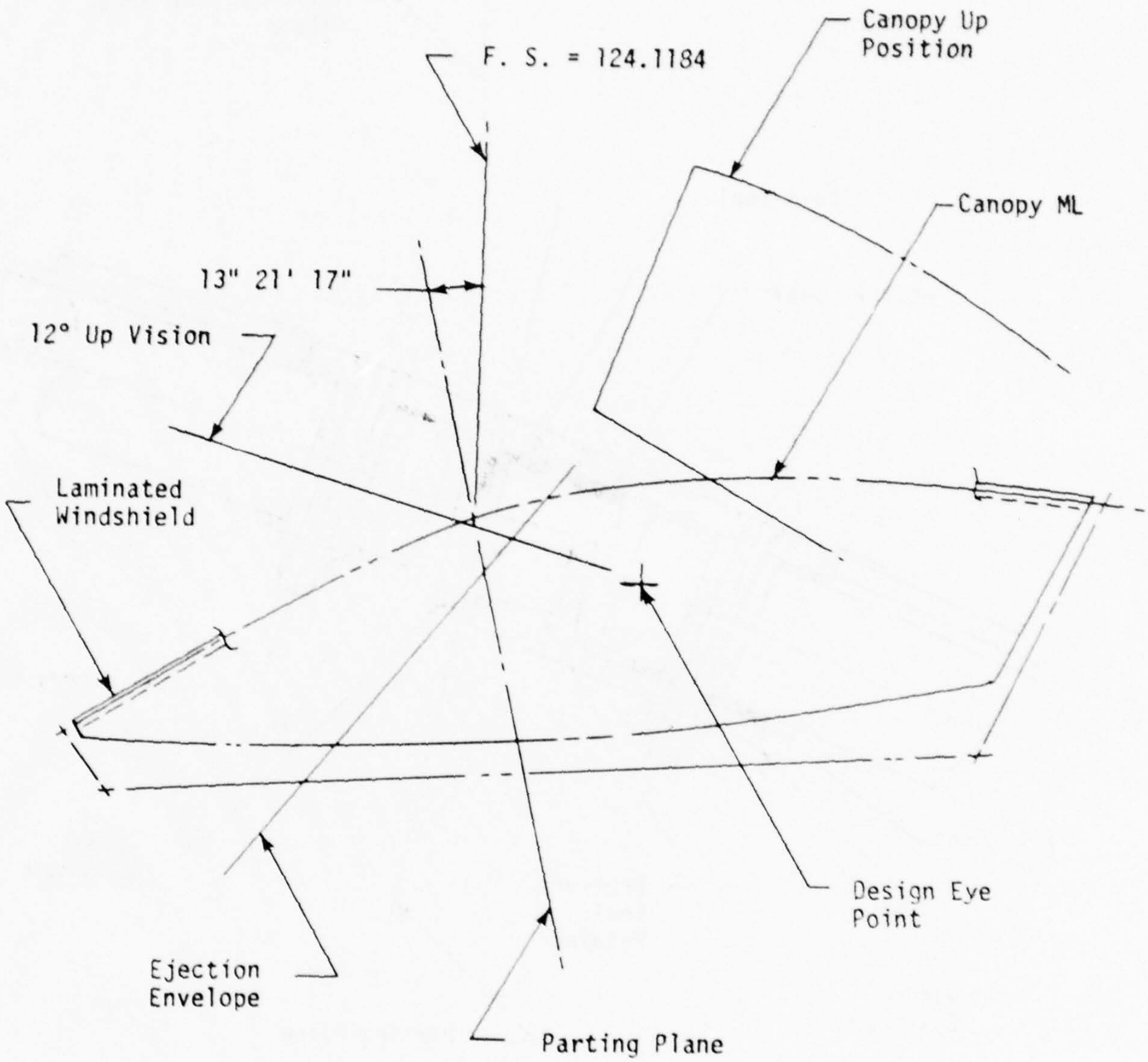


Figure 99. Parting Plane Location For Fixed Windshield Concept.

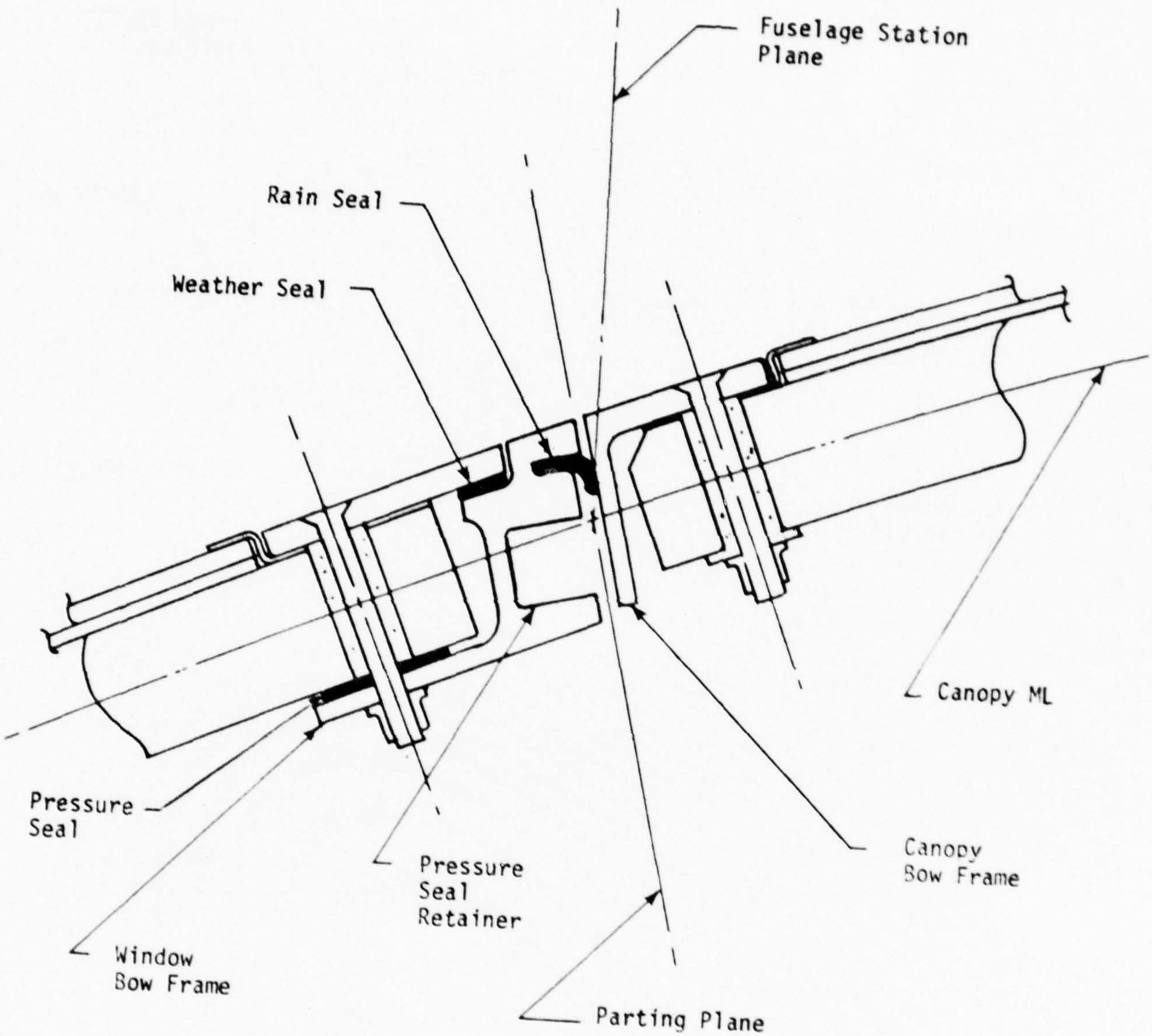


Figure 100. Bow Frame Interface For Fixed Windshield Concept.

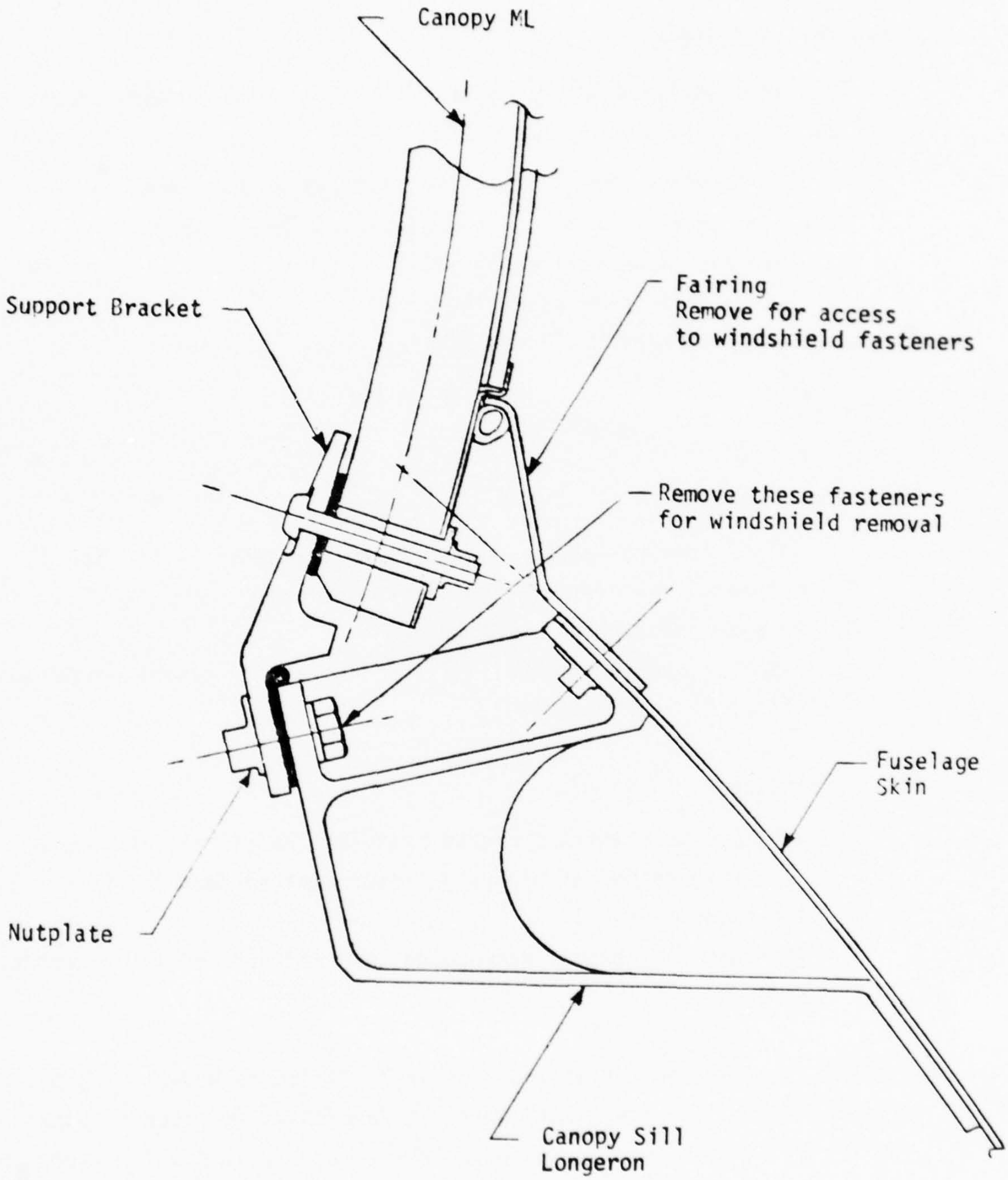


Figure 101. Typical Section Under Windshield.

Structure Redesign

This area includes the redesign of the following primary structure as well as the canopy support structure:

1. All fuselage frames under the fixed windshield area
2. Fuselage skin trim in fixed windshield area
3. Fuselage longeron (canopy sill)
4. Canopy side frame assembly
5. Canopy forward bow frame casting
6. Aft bow frame
7. Canopy side fairing

Mechanism Redesign

1. Delete forward latch. (latch 2 and 3 acceptable as is).
2. Relocate lock installation (left side only).
3. Reroute pressure and weather seals.
4. Actuator. (redesign only if the increase in canopy weight will affect its capabilities).

Canopy Jettison

1. Relocate or eliminate rocket installation (if eliminated, an alternate method of canopy jettison must be found). Figure 102 shows one method.)
2. Eliminate or reroute detonating cord and forward latch detonating cord.

The disadvantages of this design are: (1) added weight, (2) reduced pilot's vision, (3) additional parts to fabricate, (4) lack of accessibility to pilot's console for maintenance, and (5) excessive redesign to retrofit.

The advantages are: (1) the windshield can be made to withstand bird strike at higher velocity, and (2) pilot is protected from the wind blast in the event a canopy is lost in flight.

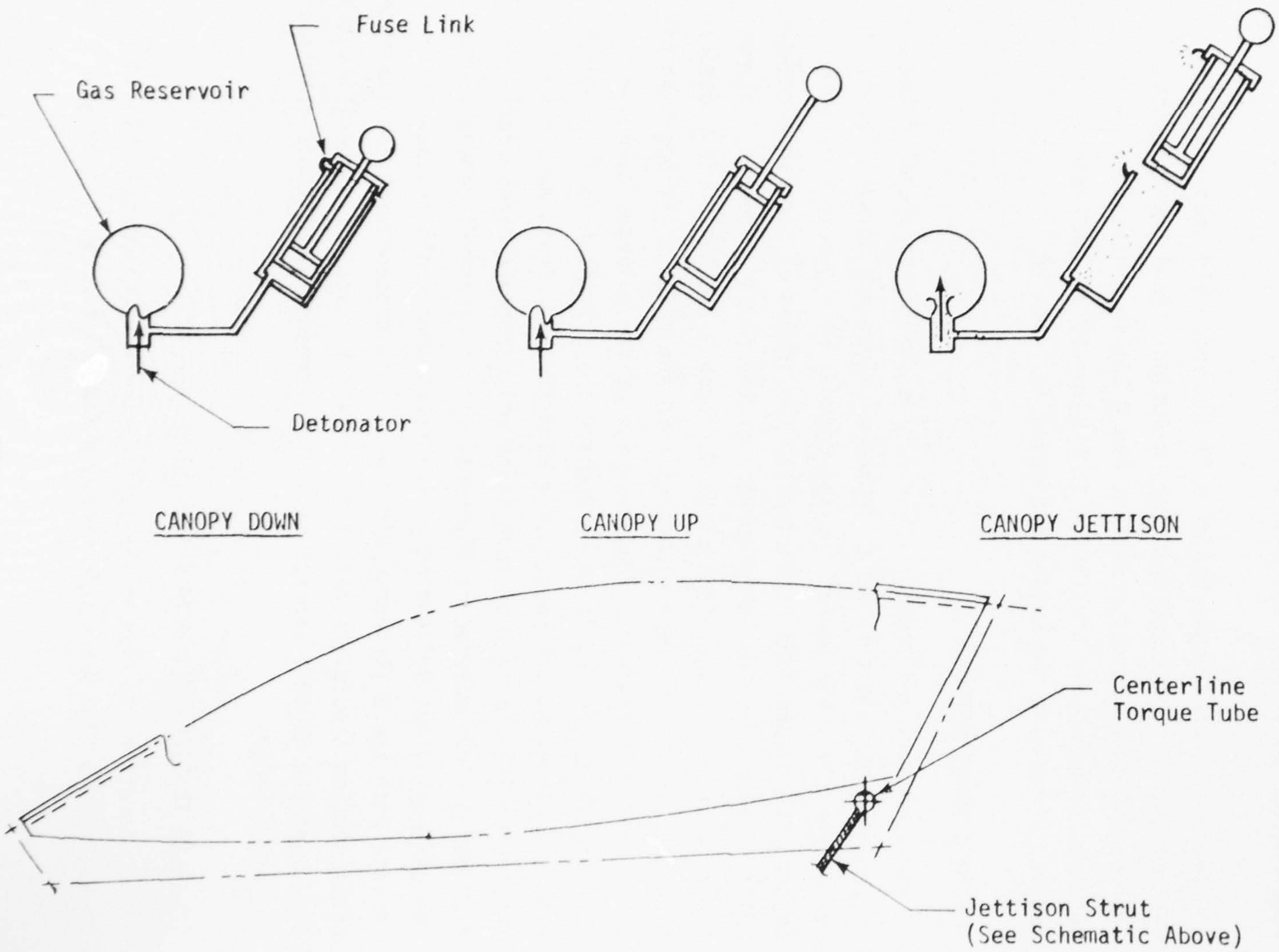


Figure 102. Canopy Jettison.

The maximum bird penetration velocity for this concept has not been established. The ratio of the transparency thickness to bow frame thickness is critical. The cross section would be as shown in Figure 86, and the polycarbonate thickness would be increased to meet the bird impact requirements at velocities up to 562 knots.

Tests have been conducted on a fixed windshield design similar to an F-15 aircraft. This configuration, a fusion bonded transparency, consisted of 0.900-inch polycarbonate and 0.100-inch as-cast acrylic. It survived a four-pound birdstrike at a velocity of 460 knots. These test results agree with the values presented in Table 26.

ENLARGED TRANSPARENCY

This recommended concept would revise the lofted shape of the canopy to provide approximately seven inches of clearance between the pilot's helmet and the inner surface of the canopy, as shown in Figure 103. The present canopy provides approximately two inches of clearance between the pilot's helmet and the inner surface of the canopy, and does not meet the requirement of the AFSC Design Handbook 2-2, DN2A1. This design handbook specifies a head clearance from the pilot's eye position of ten-inch radius minimum. The advantage of the enlarged canopy with the increased clearance would be to protect the pilot's head from contact with the canopy as the result of a bird strike. Tests have shown (Reference 16) that a 0.50-inch thick polycarbonate canopy deflected a four-pound bird impacted at 350 knots with no apparent damage to the polycarbonate, but deflection of the transparency did produce interference with the anthropomorphic head. This concept with the laminated cross section should increase the bird strike capability significantly. Disadvantages of this concept would be increased aerodynamic drag and increased weight.

ENLARGED TRANSPARENCY WITH FLAT VISION AREA

In an effort to reduce the gunfire error and increase the bird strike capability, a flat area and increased head clearance is proposed, as shown

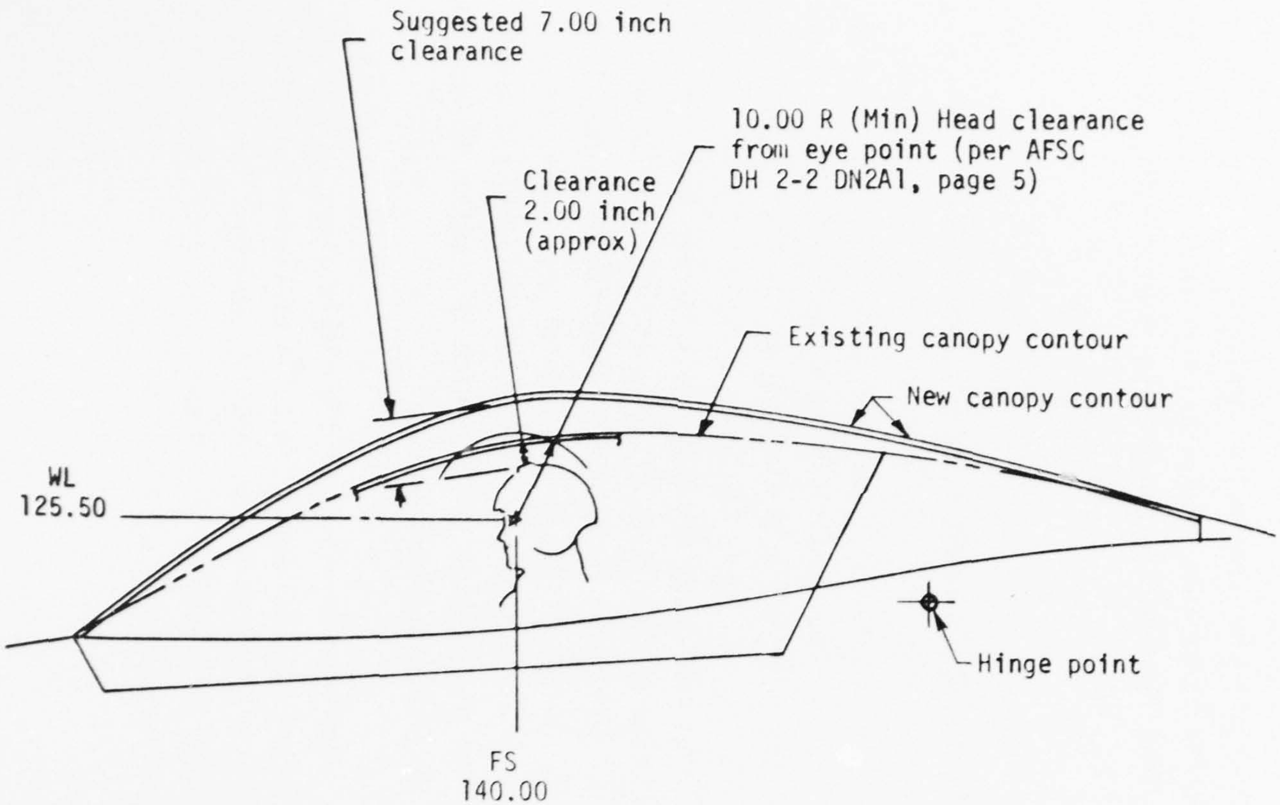


Figure 103. Canopy Design Concept Showing 7-Inch Clearance to Pilot's Helmet.

in Figure 104. This concept would meet the requirements of AFSC Design Handbook 2-2 for head clearance, and permit larger deflection of the canopy, as the result of a bird impact, without contacting the pilot's head.

The addition of the flat area would complicate the forming and possibly increase the optical distortion immediately outboard of the flat area. Additional investigation would be required to determine the extent of this distortion.

TWO PLACE COCKPIT

The application of the laminated transparency to the two place cockpit was investigated along with the single place cockpit. The results of the investigation indicated that the impact of the modified transparency on the current two place canopy substructure and its capabilities of meeting the various bird strike levels were the same as for the single place canopy.

The current two place cockpit transparency is made from a two-piece monolithic polycarbonate sheet with a splice at fuselage Station 165.00. The aft portion of the transparency was not investigated since the bird strike requirements were confined to the frontal area of the canopy. Figure 105 shows the revised splice between the laminated and monolithic transparency at fuselage Station 165.00, which is required to accommodate the thicker transparency and structure.

The emergency ejection envelope for the single place cockpit and the two place cockpit are not the same. Since the parting plane on the fixed windshield concept is dictated by the ejection envelope, the two place cockpit will have an alternate parting plane. Figure 106 shows the parting plane for the two place cockpit canopy.

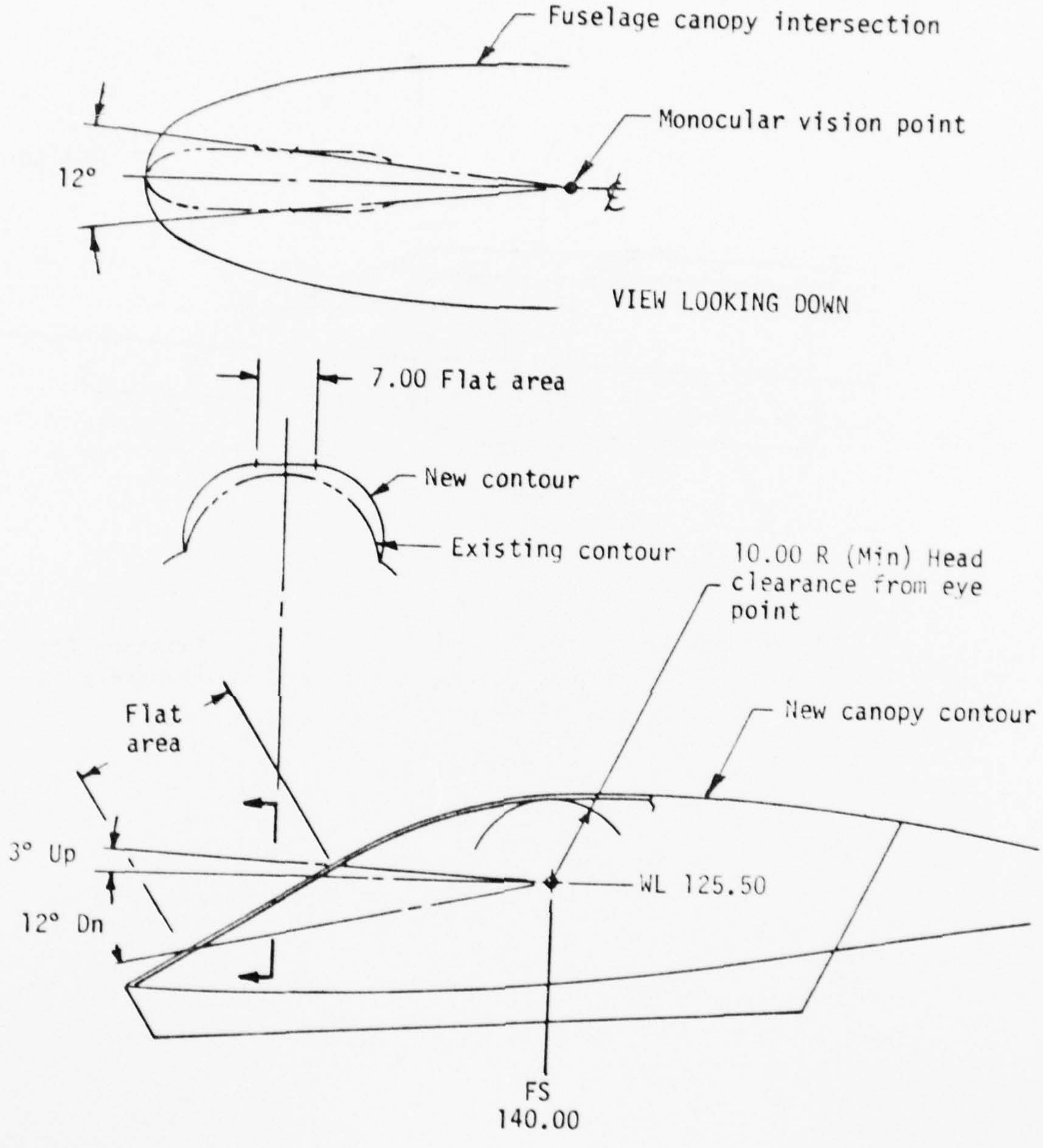


Figure 104. Canopy Design Concept With Flat Vision Area for HUD Requirement.

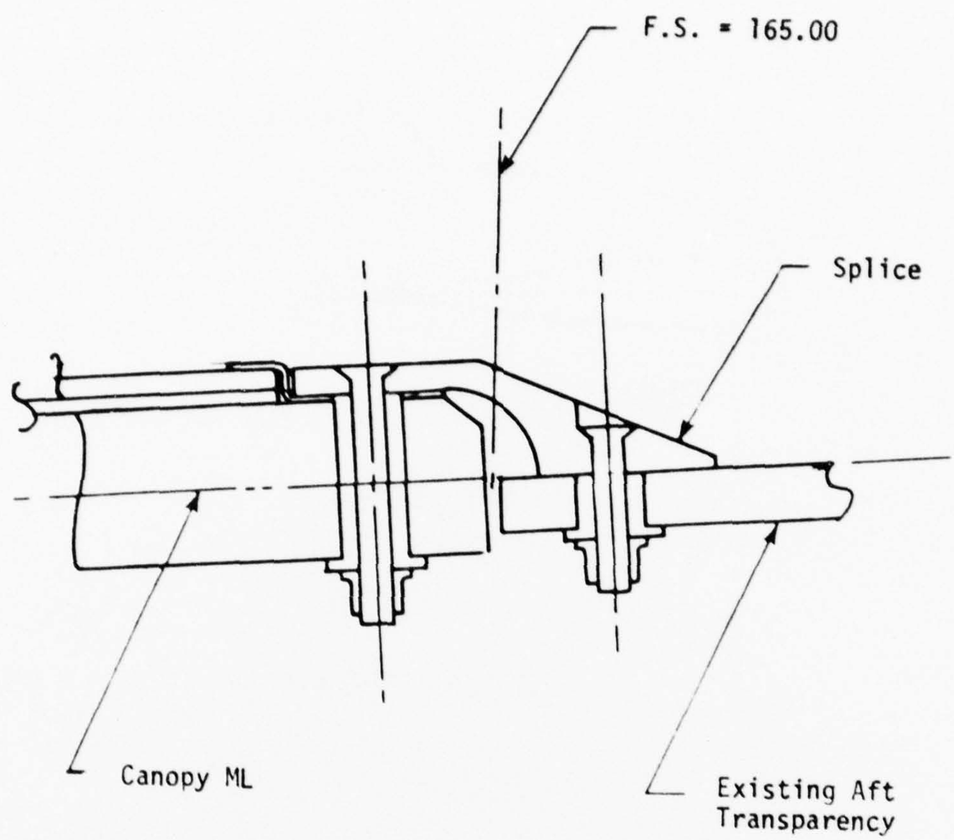


Figure 105. Transparency Splice at F.S. = 165.00.

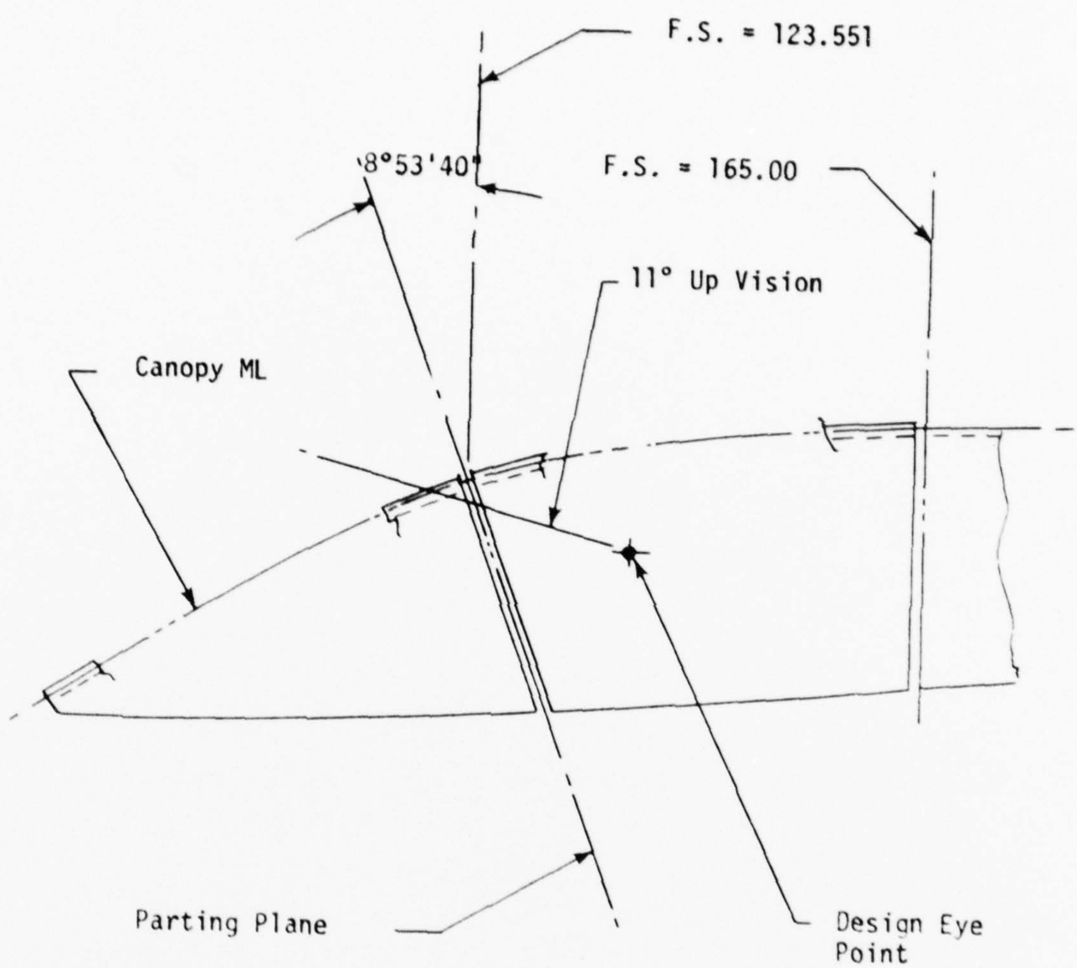


Figure 106. Parting Plane Location For F-16B.

APPENDIX
BIRD IMPACT TEST PLAN

The following plan was prepared at the beginning of the program to guide the testing to be conducted at the Von Karman Gas Dynamics Facility (VKF) located at the Arnold Engineering Development Center (AEDC), Tennessee. This test plan was based on the premise that the test canopies were capable of withstanding impact with a four-pound bird at 350 knots per the General Dynamics test results (Reference 16). The test plan was modified as shown in Section VII of this report.

TEST OBJECTIVES

The primary goals of this birdstrike test series are to provide data for validation of a dynamic math model and computer program applicable to the design of high performance aircraft canopies, to provide information required to assess the birdstrike protection afforded a pilot by the initial F-16A production canopy configuration at speeds of 350 knots by a four-pound bird, to delimit the maximum impact velocity capability of the F-16A using two and three-pound birds, to determine impact location sensitivity, and to assess if additional protection would be provided by increased thickness.

Static loading/dynamic unloading tests will be conducted to contribute to the math model validation. This type of test permits a more accurate knowledge of the loads and facilitates deflection and strain measurements.

TEST FACILITY DESCRIPTION

These tests will be conducted in the Bird Impact Test Unit (Range S3) of the Von Karman Gas Dynamics Facility (VKF), Arnold Engineering Development Center (AEDC), Tennessee, in conjunction with ARO, Inc.

The basic S3 facility is comprised of an air-operated launcher and a test stand for placement of the test panel. The launcher consists of a

31-ft-long, 8-inch-bore-diameter driver having a capacity of 10.8 cubic feet and a 60-ft-long, 7-inch-bore-diameter launch tube. The bird launch is normally accomplished by placing the bird and sabot in the launch tube immediately forward of a Mylar® diaphragm, pressurizing the driver to the desired level, and rupturing the diaphragm by means of a squib-actuated cylindrical diaphragm cutter. This diaphragm cutting technique has been shown to be consistently more reliable than the previously used dual-diaphragm venting technique. The purpose of the sabot is to adapt the bird carcass to, and prevent contact with, the bore of the launch tube during launch. The sabots will be fabricated of balsa wood having a density of approximately 10 pounds/cubic foot. The sabot is prevented from trailing the bird by a tapered stripper tube attached to the muzzle of the launch tube as shown in Figure A1.

The test area consists of a 22- by 32-foot sheltered concrete pad. Two 0.5-inch-thick steel plates spaced approximately 3 inches apart serve as the backstop; single 0.25-inch-thick steel side plates can be rolled into place for personnel protection and debris confinement.

DESCRIPTION OF TESTS

This test plan involves a maximum of 34 tests to be performed on nine coated monolithic polycarbonate transparencies, Figure A2, as shown in Table A1. The expanded canopy design programs consists of four static tests and a maximum of 29 bird impact tests on eight transparencies; the remaining bird impact test is a qualification test for the General Dynamics' F-16A canopy.

The static tests shall be accomplished by applying a known load to a point on the transparency and measuring deflection at nine points. The load shall be quickly removed to provide damping phenomena data. The bird impact tests shall be accomplished by impacting selected spots on the transparencies with birds of a known weight, at selected velocities, and at selected temperatures.

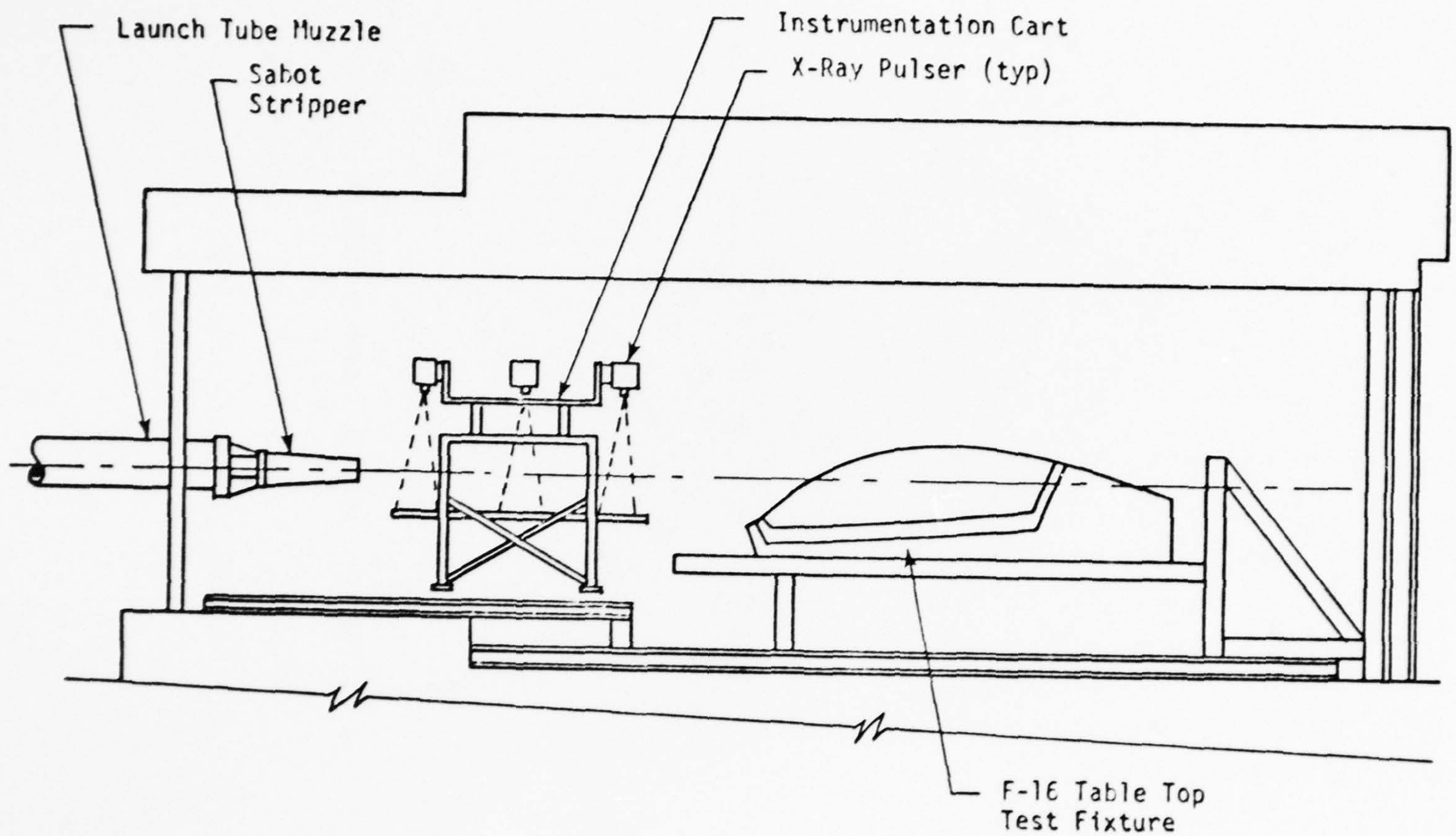
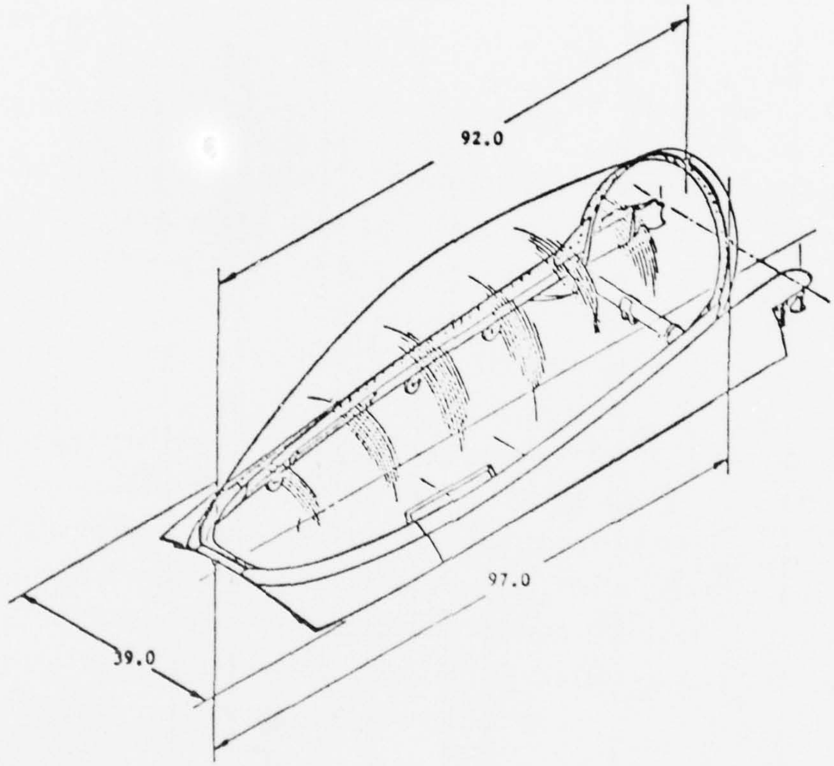


Figure A1. Test Area Arrangement.



REFERENCE: GENERAL DYNAMICS DRAWING 16K0510

Figure A2. Canopy Assembly.

TABLE A1. TEST SCHEDULE

TEST NO.	PHASE	STATIC OR BIRD	TRANSPARENCY NUMBER	t (INCHES)	SHOT LOCATION (Figure 4)	TEMP. (°F) ±10°F	STRAIN GAGE - LOCATION	V (KNOTS)	BIRD WEIGHT (POUNDS)	STATIC LOAD (POUNDS)	HUD, G.S., I.P.	PURPOSE
001	Ia	S	C1	.050	A	75	Yes - A	-	-	2500	No	Math Model
002		S	C1		A	195	Yes - A		-	2500		Math Model
003		S	C1		A	-35	Yes - A		-	2500		Math Model
004		B	C1		A	75	Yes - A	350	4	-		Math Model
005(1)		B	C1		C	75	No	350	4	-		Critical Location
006	Ib	B	C4(2)	.050	A	75	No	350	2	-	No	Penetration Velocity
007			C4		A	75	No	420	2			
008			C4		A	75	No	480	2			
009			C4		A	75	No	540	2			
010			C4		A	75	No	562	2			
011	Ic	B	C2	.050	A	195	Yes - A	350	4	-	No	Math Model
012	Id	B	C5(3)	.050	A	75	No	350	3	-	No	Penetration Velocity
013			C5		A	75	No	420	3			
014			C5		A	75	No	480	3			
015			C5		A	75	No	540	3			
016			C5		A	75	No	562	3			
017	II	B	C9	.050	-	-	-	350	4	-	Yes	Qualification Test
018	Ie	B	C3	.050	A	-35	Yes - A	350	4	-	No	Math Model
019			C3(4)		B	75	No	350	4		Yes(8)	Critical Location
020(5)			C3		D	75	No	350	4		Yes(8)	Critical Location
021	If	S	C6	.620	A	75	Yes - A	-	-	2500	No	Math Model
022(6)		B	C6		A	195	Yes - A	350	4		Yes	Math Model
023	Ig	B	C8(7)	.620	A	75	No	350	4	-	Yes	Penetration Velocity
024			C8		A	75	No	420	4		Yes(8)	
025			C8		A	75	No	480	4		Yes(8)	
026			C8		A	75	No	540	4		Yes(8)	
027			C8		A	75	No	562	4		Yes(8)	
028	Ih	B	C7(6)	.620	B	-35	Yes - B	350	4	-	Yes	Design Data
029	Ii	B	C4/C5	.500	B	-35	No	350	4	-	Yes	Optional Tests
030			C4/C5	.500	B	195	No	350	4		Yes	
031			C7/C8	.620	B	75	No	350	4		Yes	
032			C7/C8	.620	B	195	No	350	4		Yes	
033			C7/C8	.620	A	-35	No	350	4		Yes	
034			C7/C8	.620	TBD	TBD	No	350	4		Yes	

NOTES:
(1) Delete Test 005 if Test 004 fails.
(2) C1 will be used in place of C4 for Phase Ib if C1 is undamaged after Test 004 and Test 005.
(3) C2 will be used in place of C5 for Phase Id if C2 is undamaged after Test 011.
(4) C4 or C5, if available, may be used for Test 019 if Test 018 fails. Phase Ib and Id take priority over Test 019.
(5) Delete Test 020 if Test 019 fails.
(6) If C6 fails Test 022, go to Test 028. If C7 survives Test 028, use C7 for Phase Ig.
(7) C6 will be used in place of C8 for Phase Ig if C6 survives Test 022.
(8) A simulated HUD assembly is not required.

This test plan is divided into Phase I and Phase II as shown in Table A1. Phase I consists of nine sections. The AEDC/ARO shall be responsible for the test accomplishment of Phase I as detailed in the Test Procedure and Documentation sections of this test plan and for the accomplishments of Phase II as directed by the General Dynamics Test Director.

The transparencies for all tests will be attached to a canopy frame mounted on a table top test fixture, Figure A3. A second canopy frame will be available to facilitate transparency changeout after the Phase II qualification test. The table top test fixture will be supported by substructure. A heat curtain or enclosure, Figure A4, will be provided to cover the canopy assembly and table top test fixture so that the required temperatures may be met. Strain gages will be attached to Transparency Number C1, C2, C3, C6 and C7, and thermocouples will be attached to all transparencies. Appropriate data will be recorded and will include transparency surface temperatures, strain values at time, and deflection envelopes.

Phase Ia will consist of three static tests and two bird impact tests which will be performed on an instrumented transparency of 0.50-inch nominal thickness. For the static tests, 001, 002 and 003, a maximum load of 2500 pounds, or a maximum deflection of one inch, will be applied to Location A on Transparency Number C1, Figures A5 and A6, and the deflections at nine points will be recorded as shown in Figure A7. Three temperature conditions will be investigated: 75°F, 195°F, and -35°F. The temperature tolerance is ± 10°F. The cabin temperature and chamber temperature, Figure A4, will be identical. It is desirable to apply the same load for the three static tests but the potentiometers shall not be deflected more than approximately one inch. The load shall be removed quickly to provide damping data. The purpose of the static tests is to provide data to be used for verification of the dynamic math model.

The first Phase Ia bird test, 004, will impact Location A on an instrumented transparency, C1, with a four-pound bird traveling at a

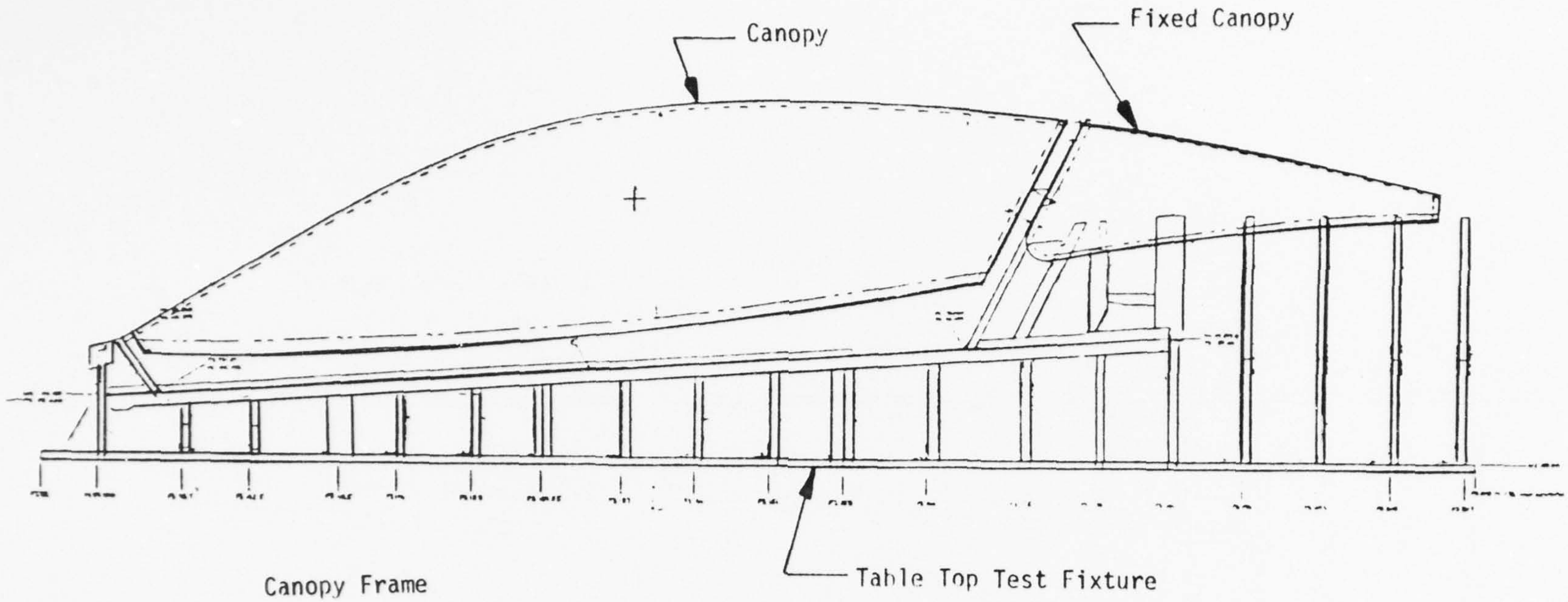


Figure A3. Canopy Assembly and Table Top Test Fixture.

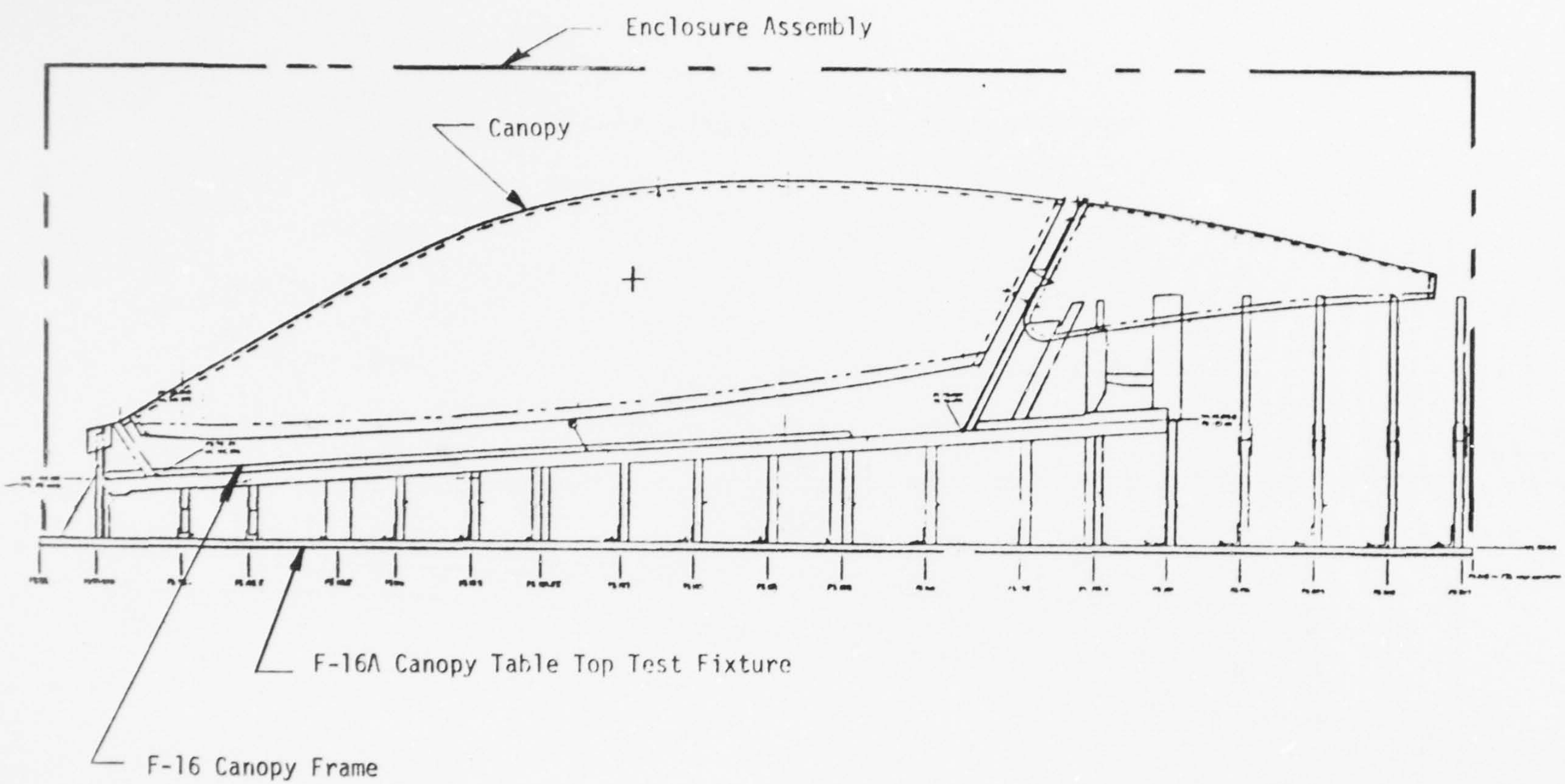


Figure A4. Heat Curtain.

AD-A070 376   DOUGLAS AIRCRAFT CO LONG BEACH CALIF   F/G 1/3
WINDSHIELD TECHNOLOGY DEMONSTRATOR PROGRAM-CANOPY DETAIL DESIGN--ETC(U)
SEP 78 M J COKER, J B HOFFMAN, J H LAWRENCE   F33615-75-C-3105
UNCLASSIFIED   MDC-J7176   AFFDL-TR-78-114   NL

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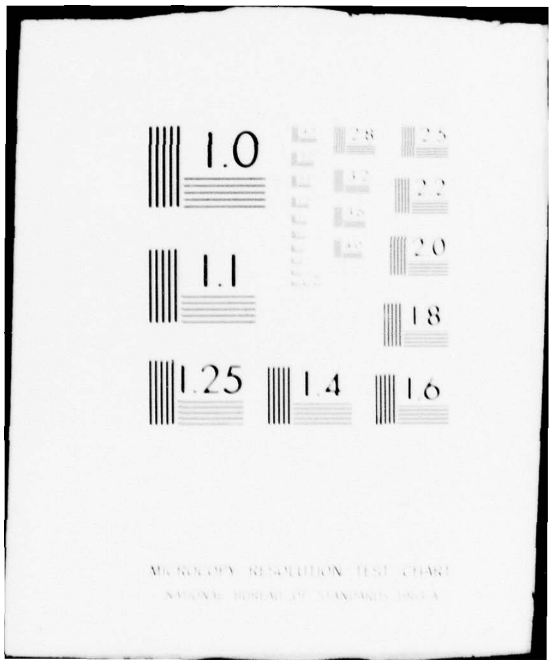
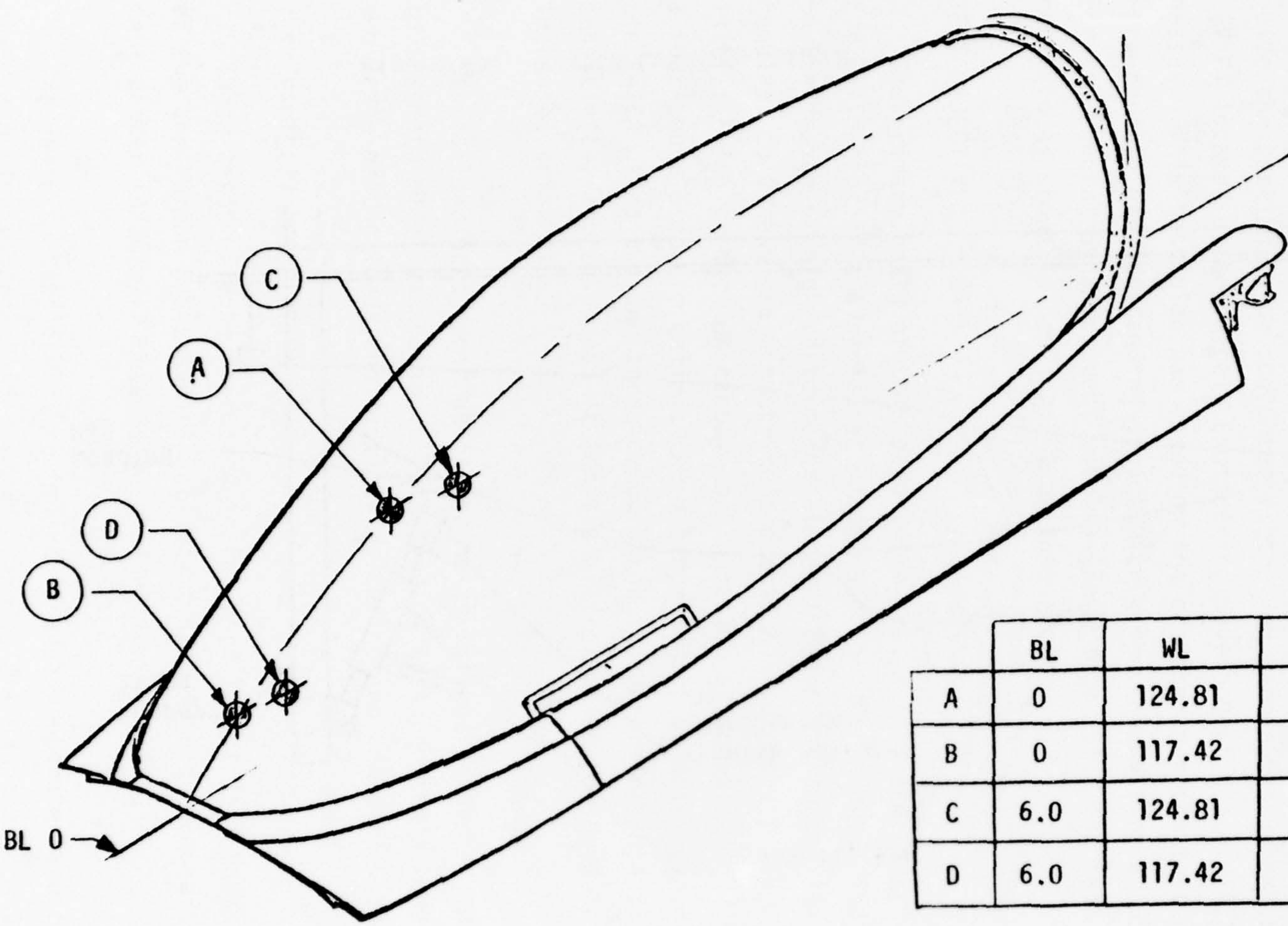

1.0
2.8
2.5
2.2
2.0
1.1
1.8
1.25
1.4
1.6
MICROCOPY RESOLUTION TEST CHART



	BL	WL	FS
A	0	124.81	112.50
B	0	117.42	100.00
C	6.0	124.81	115.16
D	6.0	117.42	102.75

Figure A5. Location of Bird Impact Coordinates.

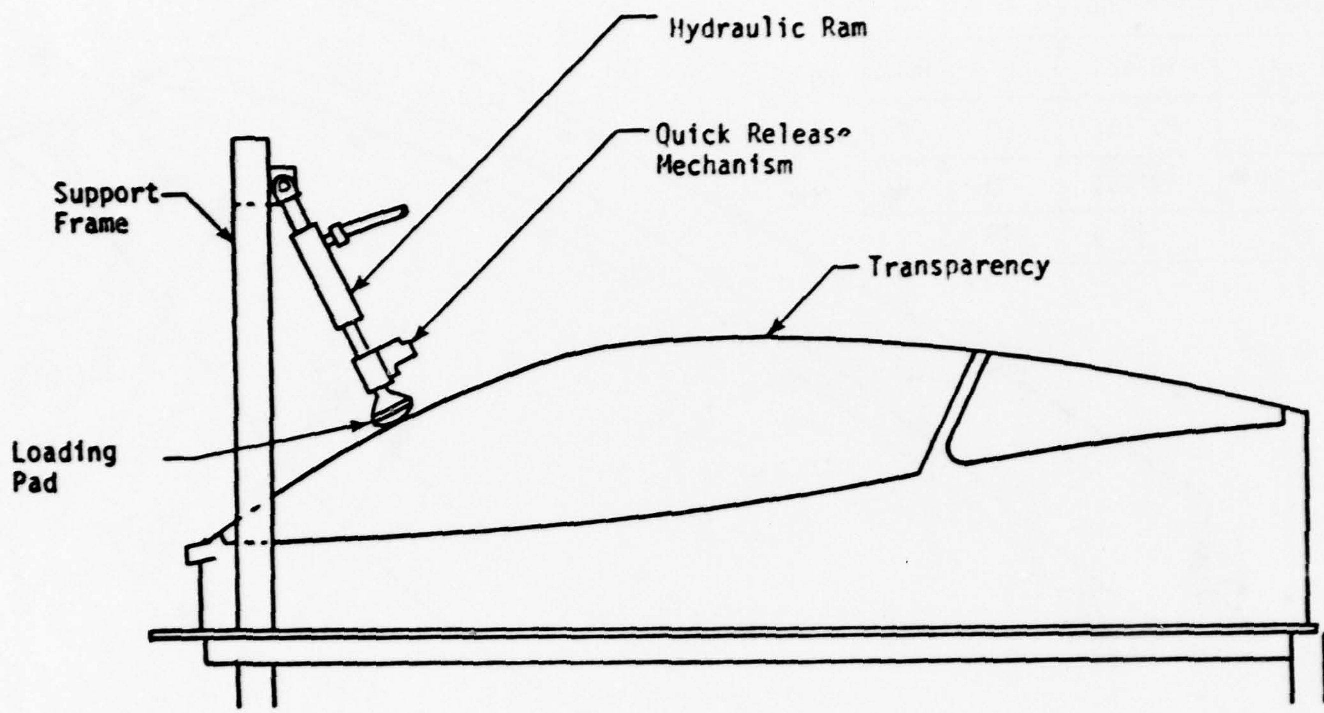
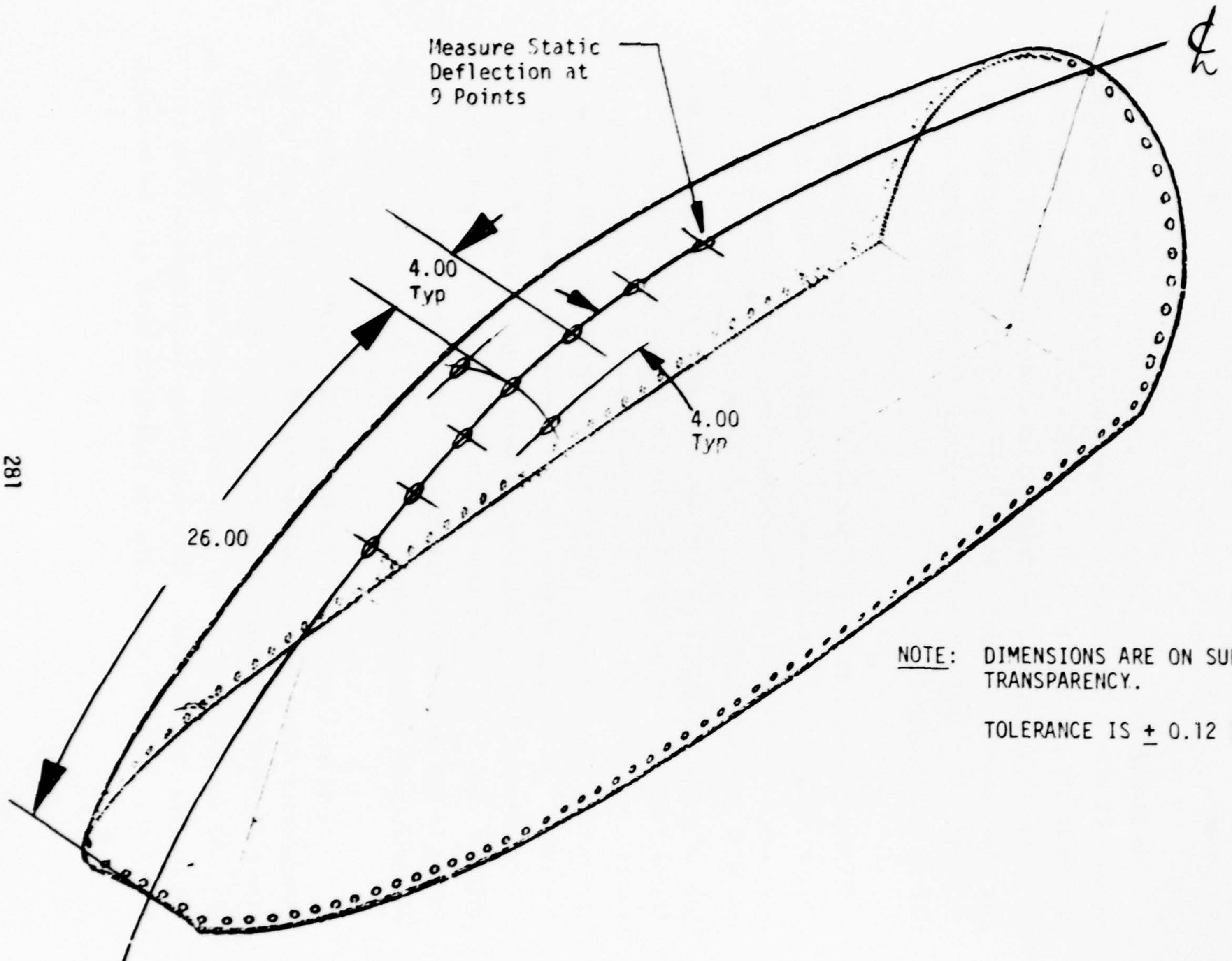


Figure A6. Static Loading Setup.



NOTE: DIMENSIONS ARE ON SURFACE OF TRANSPARENCY.

TOLERANCE IS ± 0.12 INCH

Figure A7. Static Deflection Requirements.

velocity of 350 knots; the cabin air and chamber air, Figure A4, will be held at a temperature of 75 ± 10°F. Temperature data, strain data, and the deflection curve will be recorded for the purpose of verifying the math model.

The second Phase Ia bird test, 005, will impact Location C on Transparency C1 with a four-pound bird traveling at a velocity of 350 knots; the cabin air and chamber air will be held at 75 ± 10°F. The purpose of this test is to make a comparison between a high centerline impact and a high non-centerline impact. Temperature and deflection data will be recorded. This test will not be conducted if bird penetration occurs during Test Number 004.

Phase Ib will consist of a maximum of five bird impact tests, 006 through 010, and will be conducted on Transparency Number C4, of 0.50-inch nominal thickness, with two-pound birds. The cabin air and chamber air will be maintained at 75 ± 10°F. The velocities will be 350, 420, 480, 540, and 562 knots, and the test will terminate at 562 knots or transparency failure as determined by the Douglas Test Director. The impact point will be Location A but may be varied at the discretion of the Douglas Test Director. Temperature and deflection data will be recorded. The purpose of this test is to determine the penetration velocity for a two-pound bird. Transparency Number C1 may be used for this test to conserve time if C1 is undamaged after Test Number 005; Transparency Number C4 may be used later for optional tests, Phase Ii, at the discretion of the Douglas Test Director.

Phase Ic will consist of Test Number 012 through 016 and will impact Transparency Number C5, of 0.50-inch nominal thickness at Location A, Figure A5, with three-pound birds traveling at velocities of 350, 420, 480, 540 and 562 knots or until the transparency fails as determined by the Douglas Test Director. The transparency temperature will be held at 75 ± 10°F. Temperature data and the deflection curve will be recorded.

The purpose of this test is to determine the penetration velocity. The impact loading may be varied at the discretion of the Douglas Test Director. Transparency Number C2 may be used for these tests to conserve time if C2 is undamaged after Test Number 011; Transparency C5 may be used later for optional tests, Phase Ii, at the discretion of the Douglas Test Director.

Phase II, Test Number 017, will be a qualification test directed by General Dynamics on Transparency Number C9. It is imperative that this test be completed as early in the test sequence as possible so that a second canopy frame, including a glare shield and instrument panel, will be available for Phase I testing.

Phase Ie will consist of three bird impact tests on an instrumented transparency, C3, of 0.50-inch nominal thickness. Test Number 018 will impact Location A, Figure A5, with a four-pound bird traveling at a velocity of 350 knots and the transparency temperature will be -35 $\pm$ 10°F. Temperature data, strain data, and the deflection curve will be recorded for the purpose of verifying the dynamic math model. Test Numbers 019 and 020 will impact Transparency C3 at Locations B and D, respectively, with four-pound birds traveling at 350 knots and at a 75 $\pm$ 10°F transparency temperature. Temperature data and deflection curves will be recorded. The purpose of this test is to provide a comparison between a low centerline impact, a low non-centerline impact, and a high impact from previous tests. Test Number 020 will not be conducted if bird penetration occurs during Test Number 019. If bird penetration occurs during Test Number 018, Transparency Number C4 or C5 may be used for Test Number 019. However, completion of Phase Ib and Id will take precedence over Test 019. An aircraft production type glare shield and instrument panel must be installed for Test Numbers 019 and 020.

Phase If will consist of one static test and one bird impact test on an instrumented transparency, C6, of 0.62-inch nominal thickness. For

the static test, 021, it will be necessary to apply a maximum load of 2500 pounds, or a maximum deflection of one inch, to Location A, Figures A5 and A6, on Transparency Number C6, and to record the transparency deflections at nine points as shown in Figure A7 with the transparency temperature at 75 ± 10°F. The load shall be removed quickly to provide damping data. The purpose of this test is to provide data for the math model. Test Number 022 requires C6 to be impacted at Location A, Figure A5, with a four-pound bird traveling at a velocity of 350 knots and at a 195 ± 10°F transparency temperature. Temperature data, strain data, and the deflection curve will be recorded for the purpose of verifying the dynamic math model. A simulated HUD assembly, consisting of side frames and glass, a glare shield, and an instrument panel will be installed for this test.

Phase Ig will consist of five tests, Numbers 023 through 027, and will impact a transparency, C8, of 0.62-inch nominal thickness, at Location A, Figure A5 with three-pound birds traveling at velocities of 350, 420, 480, 540, and 562 knots or until the transparency fails as determined by the Douglas Test Director. The transparency temperature will be 75 ± 10°F. Temperature data and the deflection curve will be recorded. A glare shield and instrument panel will be installed for these tests. A simulated HUD, side frames and glass, will be installed for test Number 023. The purpose of this test is to determine the transparency penetration velocity. The impact location may be varied at the discretion of the Douglas Test Director. Transparency C6 may be used for these tests to conserve time if C6 is undamaged after Test Number 022; Transparency C8 may be used later for optional tests, Phase Ii, at the discretion of the Douglas Test Director. If transparency Number C6 fails during Test Number 022, Phase Ig may be bypassed temporarily, and Test Number 028, Phase Ih, may be performed on Transparency Number C7. If C7 is undamaged, it may be used for Phase Ig tests, Test Numbers 023 through 027, and Transparency Number C8 may be used later for optional tests, Phase Ii, at the discretion of the Douglas Test Director.

Phase Ih will consist of one test, 028, and will impact an instrumented transparency, C7, of 0.62-inch nominal thickness, at Location B with a four-

pound bird traveling at 350 knots. The transparency temperature will be held at -35 $\pm$ 10°F. Temperature data, strain data, and the deflection curve will be recorded to provide information for the alternate canopy design. This transparency may be used for optional tests, Phase Ii, if it is undamaged. A simulated HUD, consisting of side frames and glass, a glare shield, and an instrument panel, will be installed for this test.

Phase Ii will consist of six optional tests and will be performed on Transparencies C4, C5, C7, and/or C8 at the discretion of the Douglas Test Director.

TEST ARTICLE DESCRIPTION

Transparency

Six of the transparencies will be 0.50-inch thick and three will be 0.625-inch thick. The thicknesses are nominal dimensions. The transparencies will be manufactured by Texstar per General Dynamics' drawings, and will be of coated monolithic polycarbonate. The specimens will be identified by part number and vendor serial number. General Dynamics will furnish thickness measurements at fifteen points for all transparencies as shown in Figure A8. The protective coating will be removed from the transparency locally to accommodate the strain gages.

The transparencies shall be used during the two phases of the program as follows:

Phase I: Five nominal 0.50-inch thick and three nominal 0.625-inch thick transparencies will be used during the expanded canopy design program as noted in Table A1. Optical rejects may be used for the five 0.50-inch thick transparencies, but the coating must be removed from these areas where strain gages will be installed.

Phase II: One nominal 0.50-inch thick transparency will be used for the qualification shot.

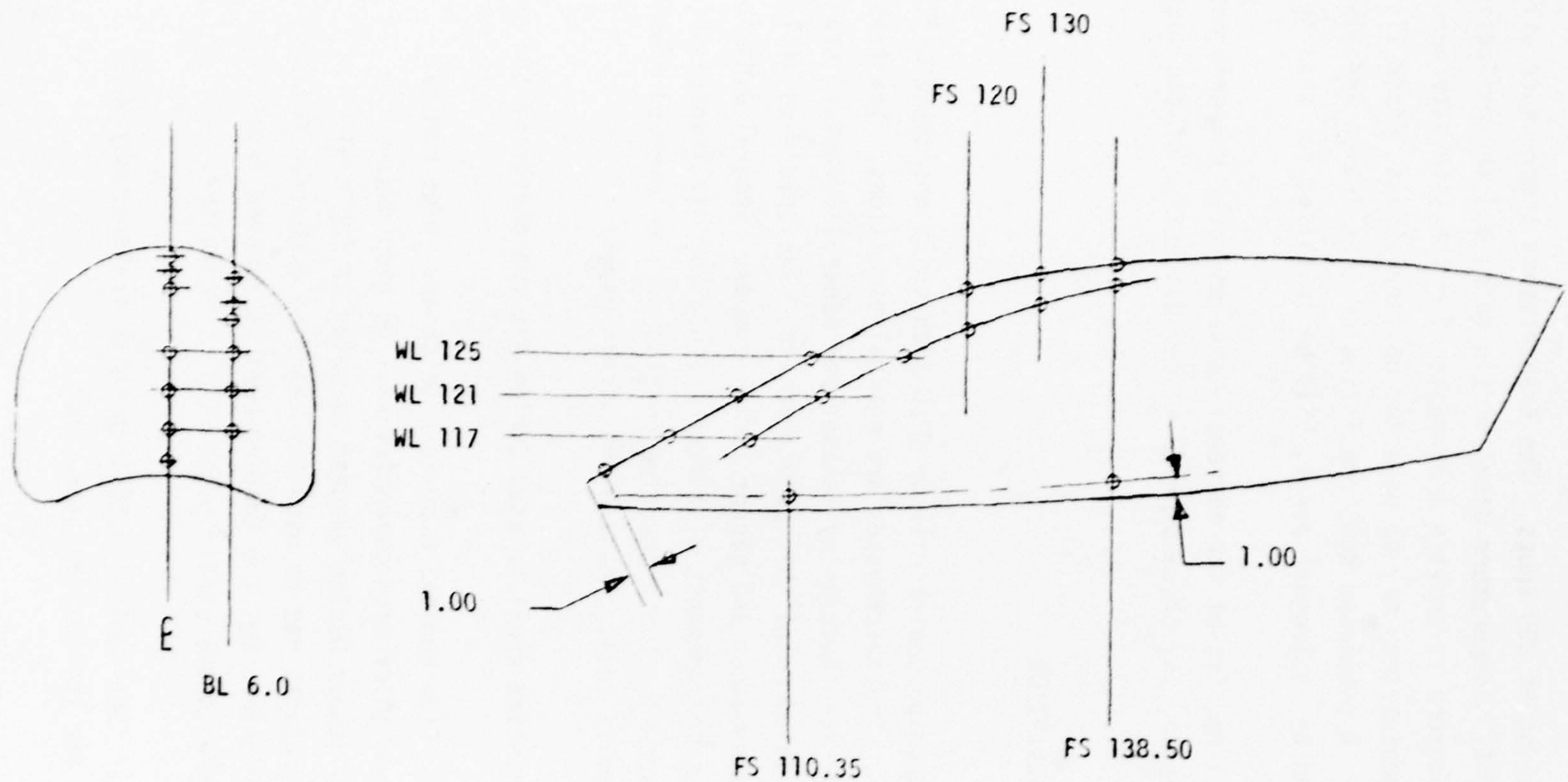


Figure A8. Transparency Thickness Measurements Requirements.

Support Structure

The Aeronautical Systems Division (ASD/YP) shall furnish a canopy frame to facilitate changeout and shall supply a modified F-16A table top test fixture (see Figure A3). The F-16A canopy table top test fixture will be modified by General Dynamics to provide an access port for interior AEDC instrumentation cables, to provide holes for installation of the anthropomorphic dummy to be provided by the 6570th Aerospace Medical Research Laboratory (AMRL), to provide a photography port, to provide thermal conditioning inlets in conjunction with AEDC/ARO, to provide accommodations for the AEDC/ARO furnished hear curtain, and to provide attachment holes for a simulated glare shield and instrument panel. ASD/YP shall furnish a second canopy frame after the qualification test is completed.

AEDC/ARO shall design and fabricate as required sufficient structure for installation of the F-16A canopy table top test fixture substructure. AEDC/ARO may drill/tap the web cross members of the table top test fixture to attach it to the substructure and bird gun platform.

Transparency Installation

General Dynamics shall install the appropriate transparency in a canopy frame using production hardware and techniques and mount it on the F-16 canopy table top test fixture as required in this test plan. AEDC/ARO and General Dynamics shall accomplish any required repairs during the test period.

Special Equipment

The 6570th Aerospace Medical Research Laboratory (AMRL) shall furnish an anthropomorphic dummy for the qualification test.

AEDC/ARO shall furnish a heat curtain or enclosure similar to ARO Drawing Number VS110476 to be used for static tests and bird impact tests (Figure A4).

AEDC/ARO shall furnish a system capable of delivering 2500 pounds of force to a transparency or deflecting the transparency a minimum of one inch, a quick release mechanism and loading pad (Douglas Drawing Number Z3942637) to enable dynamic unloading, nine deflectometers for measuring deflection, and a deflectometer support.

General Dynamics shall design, fabricate and install a readily removable simulated HUD system using the two side frames, the combining glass (simulated) of the optical gun sight, a simulated flate shield with appropriate substructure, and a simulated instrument panel.

AEDC/ARO shall design and fabricate a simulated HUD system consisting of two side frames and glass. A minimum of three sets will be required.

RESPONSIBILITIES

It will be the responsibility of Douglas Aircraft Company (DAC) to ensure that the tests are conducted in an acceptable manner, to provide the necessary detail information, to coordinate the test plan through the Air Force Flight Dynamics Laboratory (AFFDL/FEW) with the Aeronautical Systems Division (ASD/YP), the 6570th Aerospace Medical Research Laboratory (6570th AMRL), the Arnold Engineering Development Center (AEDC/ARO), the Air Force Materials Laboratory (AFML), and the General Dynamics Company, and to gather engineering data which will form a data base for the expanded canopy design effort and for application to the validation of the dynamic math model and computer program.

The transparent hi-visibility canopies are part of a canopy assembly designed by General Dynamics. It will be the responsibility of the ASD/YP to furnish the F-16 canopy table top test fixture to support the canopy assembly, eight transparencies and one canopy frame for the expanded canopy design program, one transparency and canopy frame for the F-16A qualification test, and one simulated HUD system to be used for the qualification shot and which may be used for subsequent shots. Final

disposition of the test hardware, including canopies, will be directed by ASD/YP. The General Dynamics' qualification test will be controlled by the ASD/YP.

It will be the responsibility of the 6570th AMRL to provide an instrumented anthropomorphic dummy and to evaluate the pilot injury potential if the canopy strikes the pilot's helmet during the birdstrike event. This evaluation will be coordinated through the ASD/YP for the qualification test and the AFFDL/FEW for all other tests.

It will be the responsibility of the AEDC/ARO to design, fabricate and install a rigid table top substructure, to furnish the target platform to support the table top test fixture, and to drill/tap the web cross members of the table top fixture to attach it to the substructure and bird gun platform is necessary.

The AEDC/ARO will furnish a heat curtain or enclosure similar to ARO Drawing Number 110476 or an equivalent system and it may be necessary to modify the table top test fixture to be compatible with the heat curtain and to enclose the inboard surface of the canopy to meet temperature requirements. Proposed configuration changes to the table top test fixture will be reviewed by the on-site General Dynamics' engineer and all changes will be documented and will require authorization by the ASD/YP. The AEDC/ARO will furnish thermal conditioning equipment, support facilities, including office space and office equipment, the bird gun and birds, and instrumentation. The test transparencies will be instrumented by the AEDC/ARO with strain gages and thermocouples for data collection to be used for the bird impact math model in accordance with the requirements of this test plan. Cameras and film will be provided for deflection measurements and overall photo coverage. Test engineer, technicians and cameramen will be provided by the AEDC/ARO.

The AEDC/ARO will be responsible for the test accomplishments as detailed in the Test Procedure and Documentation Sections of this test plan and will provide copies of data to all activities in accordance with a data distribution plan to be coordinated through the AFFDL/FEW.

Five days after the first test and once each week thereafter, the following data will be sent by the AEDC/ARO to Douglas: The oscillograph strain gage data, the temperature data, the deflection versus time plots, and the deflection IBM cards. Ten days after the first test and once each week thereafter, the digitalized strain data (strain versus time plots and IBM printout), and deflection films shall be sent to Douglas.

The AEDC/ARO will set up a temperature recorder, prepare a strain gage data sheet, and provide Douglas with a measurement system schematic.

It will be the responsibility of General Dynamics to accomplish canopy changeouts between Phase I tests, accomplish on-site canopy frame and table top test fixture repairs if and when necessary, to drill and tap holes where required to install the 6570th AMRL head/neck form, to provide an access port in the table top test fixture for interior AEDC/ARO instrumentation cables if required, to add a sealed viewing port at the direction of the AEDC/ARO in the bottom of the table top fixture to permit photographic documentation of the deflection of the inside surface of the canopy during bird impact at selected temperatures, and to modify the table top fixture to be compatible with the AEDC/ARO thermal conditioning equipment. General Dynamics will be responsible for engineering the qualification shot (preparing the test plan, installing the instrumentation, reducing the data, and writing the final report) with guidance assistance being provided by the AFFDL/FEW, the 6570th AMRL, Douglas, and the AEDC/ARO. General Dynamics will be responsible for the design, fabrication, and installation of the readily removable, simulated HUD system consisting of the two side frames and the combining glass (simulated) of the optical gun sight, and a simulated flare shield and instrument panel. This system will be used for the qualification test, Phase II, and may be used for subsequent tests in Phase I. General Dynamics will be responsible for providing transparency

thickness measurements to Douglas Aircraft for all transparencies to be tested and will assure that the transparencies are uncoated in the areas where strain gages will be installed.

The AEDC/ARO will be responsible for fabricating a simulated HUD system consisting of two side frames and a simulated combining glass to be used during Phase I tests. A minimum of three sets will be required.

TEST AND INSTRUMENTATION REQUIREMENTS

Test Procedure and Documentation

The AEDC shall be responsible for the test accomplishment as detailed in the Test Procedure and Documentation sections of this test plan.

Static Loading/Dynamic Unloading Tests

Nine deflectometers shall be mounted on the inside of the transparency to measure static deflection at the nine locations shown in Figure A7. The deflectometers must not impinge on the strain gages or grid pattern, must be capable of deflecting at least two inches before bottoming out, and shall be connected to an oscillograph recorder. A hydraulic loading system capable of delivering 2500 pounds or deflecting the transparency one inch shall be located to apply a static force normal to the transparency at Location A, Figure A5. A quick release mechanism and loading pad to enable a dynamic unloading of the transparency for calibration purposes shall be provided as shown in Figure A6. The test setup shall have the capability of maintaining temperatures of 195°F and -35°F. The deflectometer locations and angles, and the coordinates of the nine deflection points, will be documented on Data Sheets 15 and 16, noted at the end of the test plan.

Bird Weight Requirements

The AEDC/ARO shall provide appropriately packaged chickens as required in Table A1. The test birds shall be either freshly killed or quickly frozen after killing and slowly thawed for 24 hours prior to

testing. Each test bird shall consist of a complete carcass weighing $2.0^{+.2}_{-0}$ pounds, $3.0^{+.2}_{-0}$ pounds or $4.0^{+.2}_{-0}$ pounds. Weight adjustments may be made by clipping carcass appendages or by injecting water into the body cavity.

Bird Velocity Requirements

The AEDC/ARO shall provide a bird gun with an X-ray bird velocity measurement system capable of meeting the velocity requirements of Table A1.

The bird velocity tolerance shall not exceed the following values:

KNOTS	FT./SEC.	MPH.
350 ± 2.5%	591 ± 2.5%	403 ± 2.5%
420 ± 2.5%	709 ± 2.5%	484 ± 2.5%
460 ± 2.5%	777 ± 2.5%	528 ± 2.5%
540 ± 2.5%	912 ± 2.5%	621 ± 2.5%
562 ± 2.5%	949 ± 2.5%	647 ± 2.5%

Location of Impact Point

The impact points shall be located as shown in Figure A9. The tolerance on the locations shall not exceed ± 0.12 inch. The locations and coordinates shall be recorded in Data Sheets 2 and 3.

Impact Point Tolerance

The tolerance on the bird impact points shown in Figures A5 and A9 shall not exceed ± 1.00 inch.

Thermal Requirements

The thermal requirements were chosen as the absolute minimum and maximum temperatures that could be experienced for bird impact. The hot temperatures simulate a hot atmosphere supersonic cruise followed by an emergency descent to 8,000 feet and 350 knots, followed immediately by bird impact. The cold temperatures simulate a cold atmosphere flight at 200 feet and 350 knots prior to bird impact. The temperatures shown

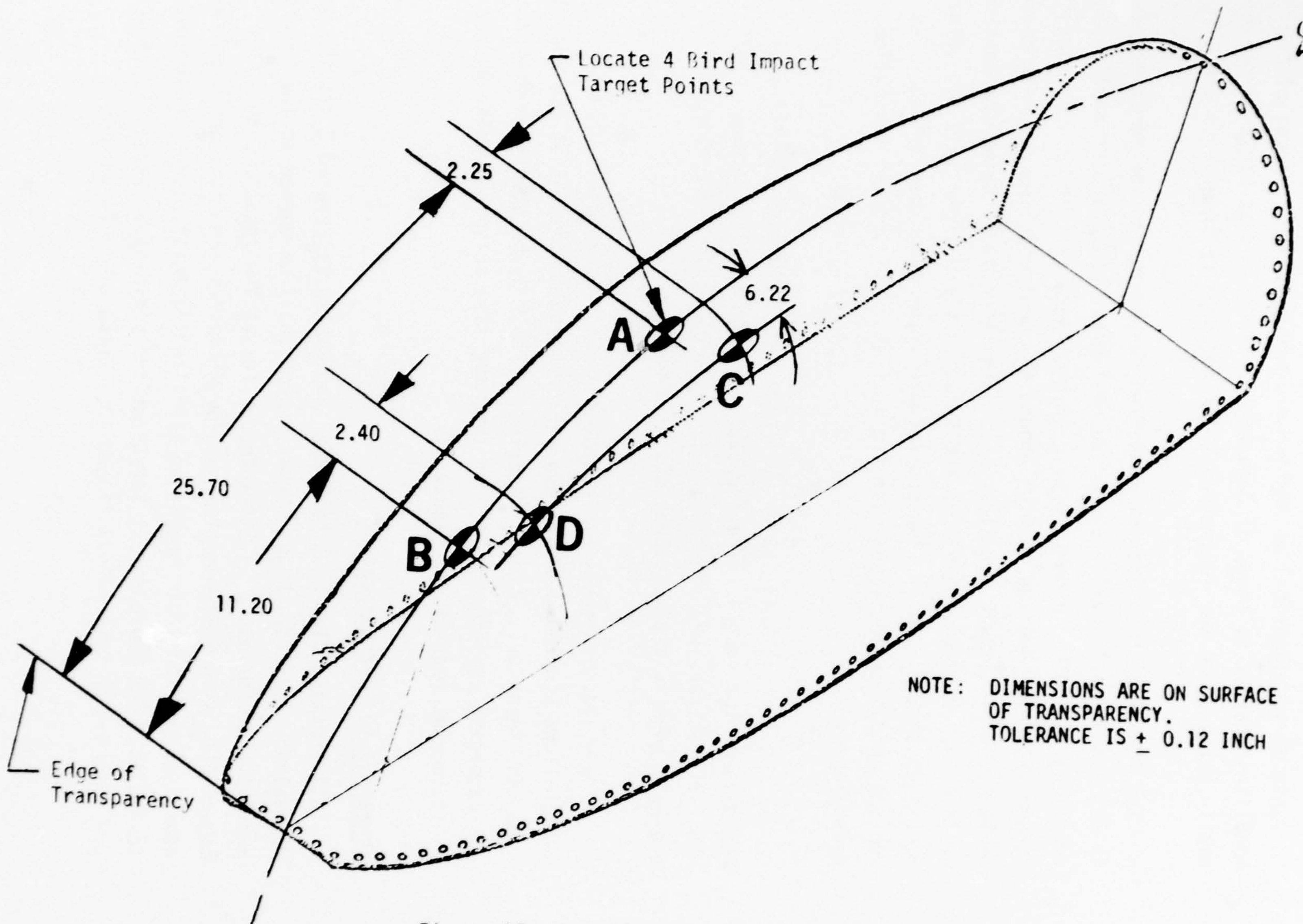


Figure A9. Location of Bird Impact Target Points.

in Table A1 are identical on the inboard and outboard surfaces of the monolithic canopy as shown in Figure A10. These are as close as can be easily obtained in the laboratory. The temperature tolerances are ± 10°F.

The AEDC/ARO shall provide a heating system to provide temperatures to 195°F and shall provide LN_2 or an equivalent material for cooling the specimens to -35°F. A heat curtain or enclosure similar to ARO Drawing Number VS110476 shall be mounted around the test specimen and F-16 canopy table top test fixture per Figure 10. The temperature range for control would be approximately -35°F to 195°F, as required from Table A1. Aforementioned ducting should flow approximately one pound per second. The ducting should be insulated with two-inch thick fiberglass insulation or equivalent.

As required, either hot or cold air shall be blown into the area enclosed by the canopy. The temperature range shall be between -35°F and 195°F as required from Table A1. It will require approximately one-half hour to attain steady state temperatures.

Before the bird tests are conducted, the temperatures for each specimen shall be monitored to achieve steady state conditions at the locations shown in Figure A11. Thermal conditions shall be constantly monitored on the recording equipment to ensure that the test setup temperatures are maintained as required per Table A1.

Thermocouple Instrumentation

All thermocouples will be copper constantan calibrated to I.S.A. or equivalent standard. Eight thermocouples shall be bonded to the interior and exterior surface of all transparencies at the locations shown in Figure A11, with GA-2 adhesive made by Micro-Measurements, or an equivalent adhesive, per the Instrumentation Installation Instructions Section of this test plan. The thermocouple locations and coordinates shall be measured and recorded on Data Sheets 11 and 12. Temperature measurements

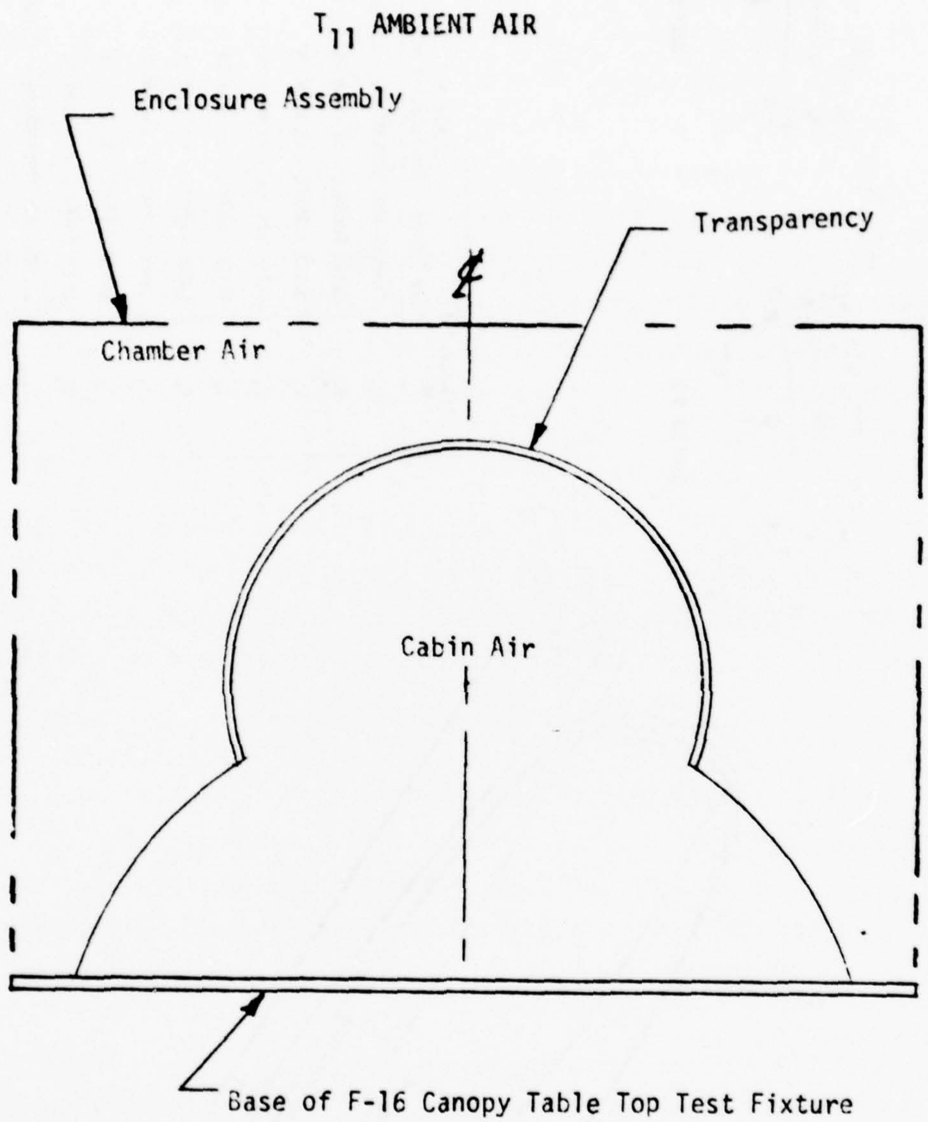
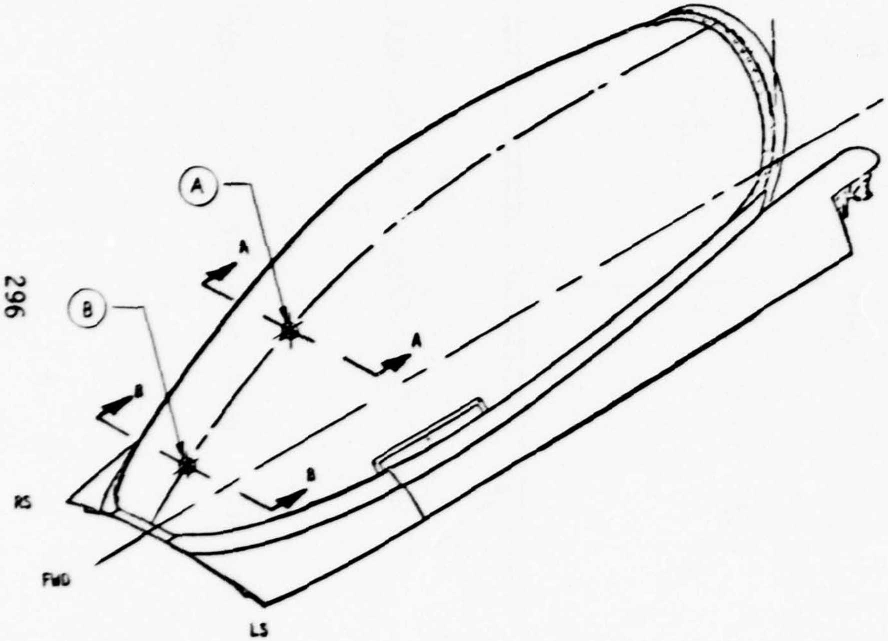
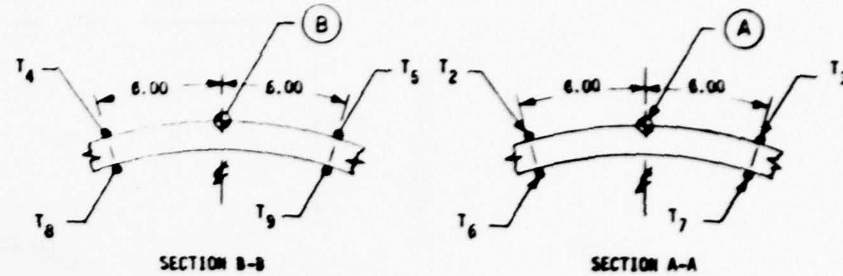


Figure A10. Temperature Definition Schematic.





THERMOCOUPLE NO.	LOCATION See Figure 4 for Location of A and B
T_1	CHAMBER AIR TEMPERATURE - 3" ABOVE 'A'
T_2	OUTER SURFACE - RS NEAR 'A'
T_3	OUTER SURFACE - LS NEAR 'A'
T_4	OUTER SURFACE - RS NEAR 'B'
T_5	OUTER SURFACE - LS NEAR 'B'
T_6	INNER SURFACE - RS NEAR 'A'
T_7	INNER SURFACE - LS NEAR 'A'
T_8	INNER SURFACE - RS NEAR 'B'
T_9	INNER SURFACE - LS NEAR 'B'
T_{10}	COCKPIT AIR TEMPERATURE - 3" BELOW 'A'
T_{11}	AMBIENT AIR TEMPERATURE

Figure A11. Thermocouple Locations.

utilizing thermocouples will also be made of the temperature chamber air, pilot's head air, and ambient outside air.

The AEDC/ARO will install thermocouples and will complete the measurement circuit as required for recording temperatures. A recorder for monitoring and recording temperatures at the rate of 12 thermocouples in 6 seconds total time between will be required. Continuous monitoring and recording will be necessary during heating and cooling and during bird impact testing for each shot. All test events must be recorded on the temperature records as they occur, such as "lights on", "fire", etc.

Strain Gage Instrumentation

Strain gage locations are shown in Figures A12 and A13, and Table A2, and will be installed on Canopies C1, C2, C3, C6 and C7, as shown in Table A1. All strain gages are single active legs and will require dummy bridge completion networks. The type of strain gage to be used is EP-08-125AV-120 (Option B64) made by Micro-Measurements. The strain gages shall be bonded to the transparency interior and exterior surface, frame, and hooks with GA-2 adhesive made by Micro-Measurements per the Instrumentation Installation Instructions section of this test plan. The strain gage locations and coordinates shall be measured and recorded on Data Sheets 8, 9 and 10.

The AEDC/ARO shall provide system wiring, completion networks, power supplies, balancing and sensitivity controls for resistance bridge measurements with outputs compatible to FM recording equipment, and signal conditioning and FM tape recorders for up to 24 data channels. All tapes must carry precision time code for data synchronization. The AEDC/ARO shall record either by FM tape of by oscillograph, strain time history measurements from the structural gages and the strain gages located on the transparency.

The strain gage instrumentation to be used must meet the following requirements and conditions:

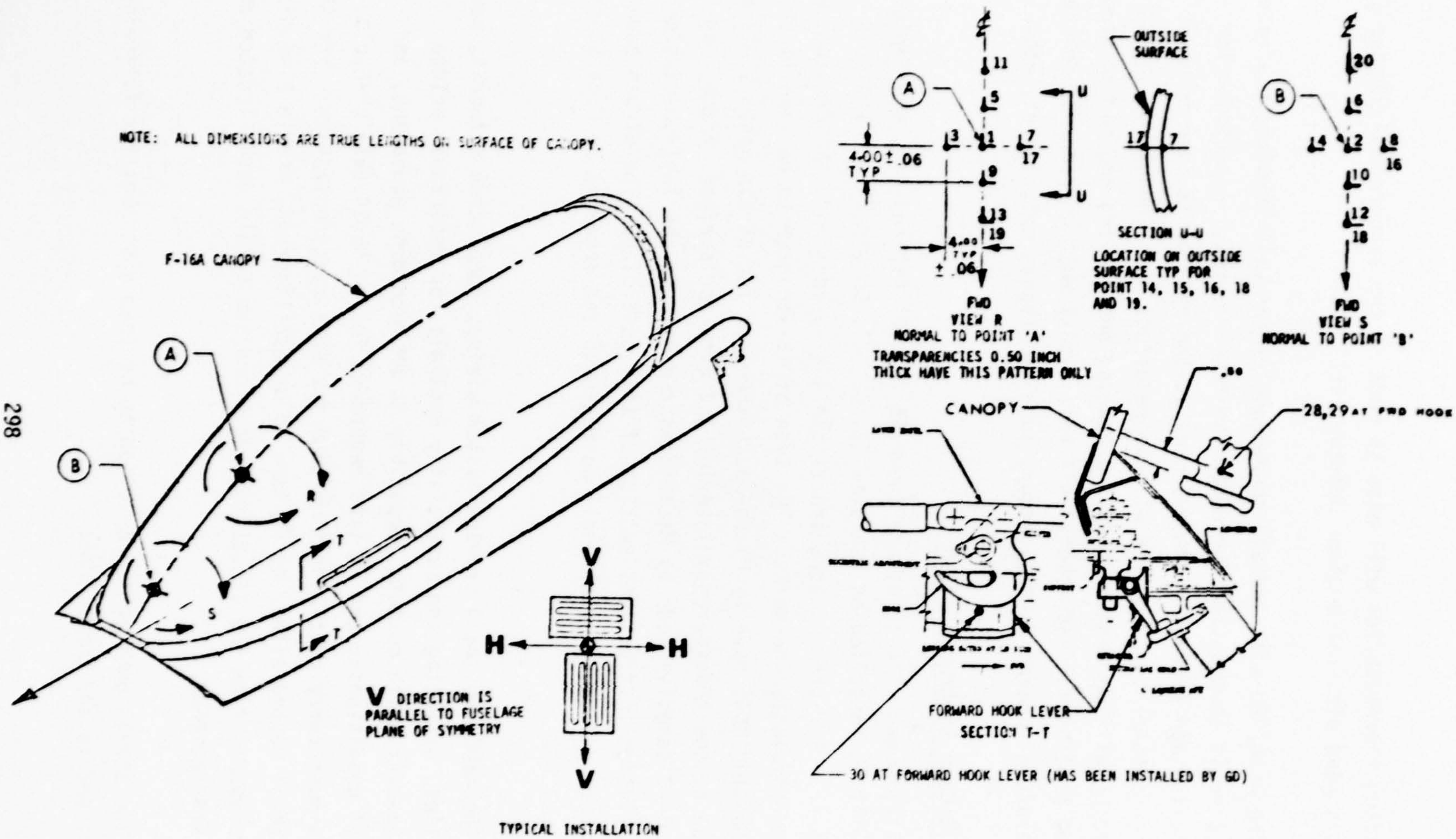


Figure A12. Strain Gage Locations.

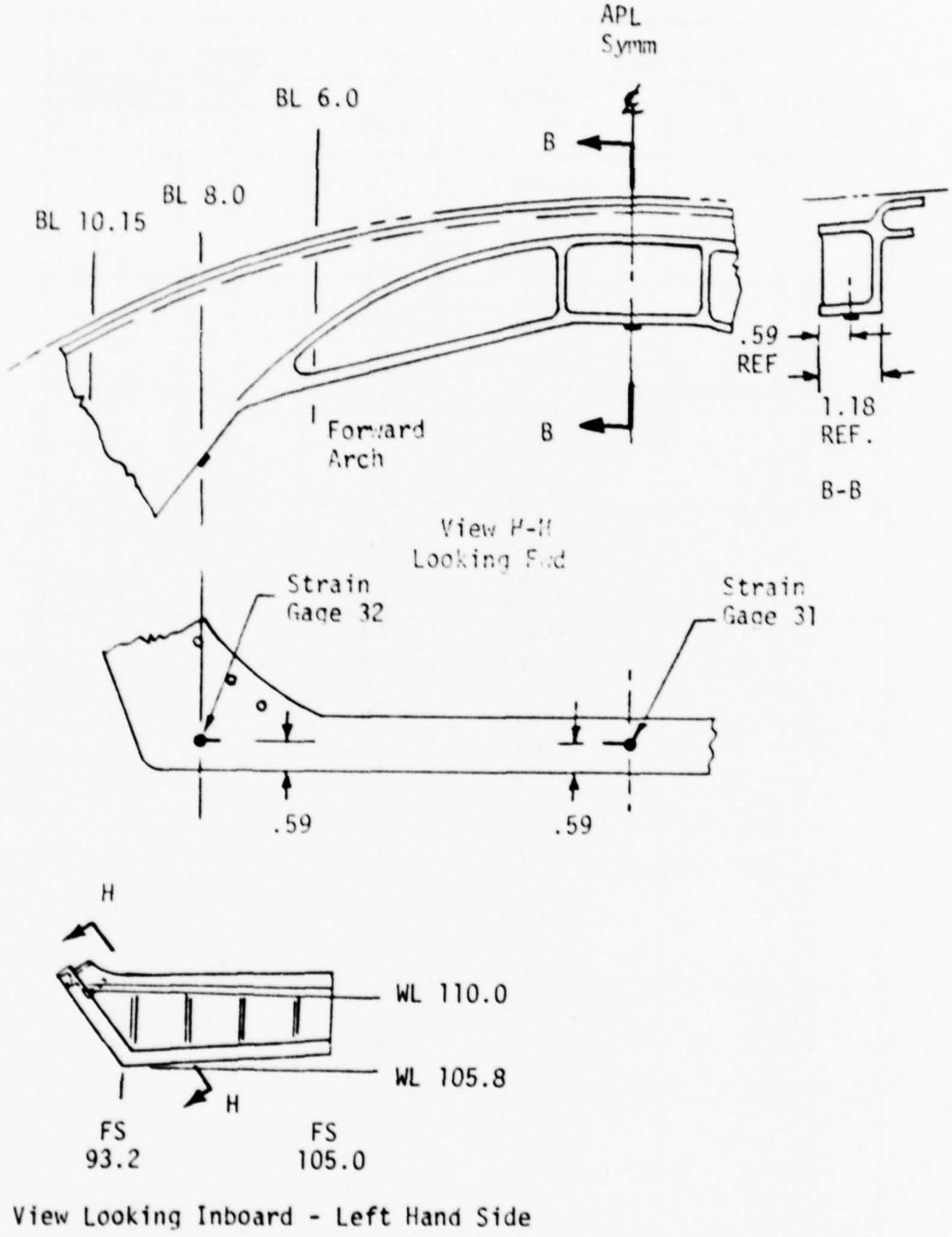


Figure A13. Strain Gage Location Forward Canopy Frame.

TABLE A2. STRAIN GAGE LOCATION AND RECORDING CHANNEL SEQUENCE NUMBERS PER SHOT LOCATION

STRAIN GAGE LOCATION			CHANNEL SEQ. NO.	
LOCATION NO. & GAGE DIR.	MOUNTED ON	SIDE	SHOT LOCATION A	SHOT LOCATION B
1 H	Canopy	In	1	
1 V	Canopy	In	2	
2 H	Canopy	In		1
2 V	Canopy	In		2
3 H	Canopy	In	3	
3 V	Canopy	In	4	
4 H	Canopy	In		3
4 V	Canopy	In		4
5 H	Canopy	In	5	
5 V	Canopy	In	6	
6 H	Canopy	In		5
6 V	Canopy	In		6
7 H	Canopy	In	7	
7 V	Canopy	In	8	
8 H	Canopy	In		7
8 V	Canopy	In		8
9 H	Canopy	In	9	
9 V	Canopy	In	10	
10 H	Canopy	In		9
10 V	Canopy	In		10
11 V	Canopy	In	11	
12 H	Canopy	In		11
12 V	Canopy	In		12
13 H	Canopy	In	12	
13 V	Canopy	In	13	
16 H	Canopy	Out		13
16 V	Canopy	Out		14
17 H	Canopy	Out	14	
17 V	Canopy	Out	15	
18 H	Canopy	Out		15
18 V	Canopy	Out		16
19 H	Canopy	Out	16	
19 V	Canopy	Out	17	
20 V	Canopy	In		17
28 H	Edge	Out	18	18
28 V	Edge	Out	19	19
29 H	Edge	In	20	20
29 V	Edge	In	21	21
30	Fwd Hook		22	22
31	Fwd Frame		23	23
32	Fwd Frame		24	24

- The motion to be measured will be a lightly damped oscillation with an expected period of about 15 milliseconds and shall be monitored and adjusted accordingly. Peak velocity will be in the region of 2,000 cm/sec, and possibly as great as 10,000 cm/sec. Peak displacements will be in the region of 10 inches.

- The primary period of interest will include the duration of impact, which will be about half a millisecond, and approximately 5 to 10 milliseconds subsequent to impact. This time period will be monitored and adjusted accordingly.

- The test method must permit these measurements to be made without interfering with the bird strike test. In other words, there must be free access to the test surface for the projectile, and the test apparatus must not affect the free movement of the test surface.

The expansion/contraction rates and creep of the specimens at various temperatures will be required. Particular emphasis must be given to the areas around the latching mechanism. Data for this purpose will be obtained from the strain gages and the thermocouples installed on the canopy and adjoining structure by recording prior to and during heating or cooling preparations.

Calibration of instruments will be required to verify that total systems requirements have been satisfied. Evidence of recent certification will be provided to the Douglas representative.

Dynamic Deflection Measurements

High speed cameras will be used to measure deflection versus time at the birdstrike event. A minimum of four high speed movie cameras and color film (5000 frames per second) are required. Suggested camera locations are shown in Figures A14 and A15. The camera locations will be

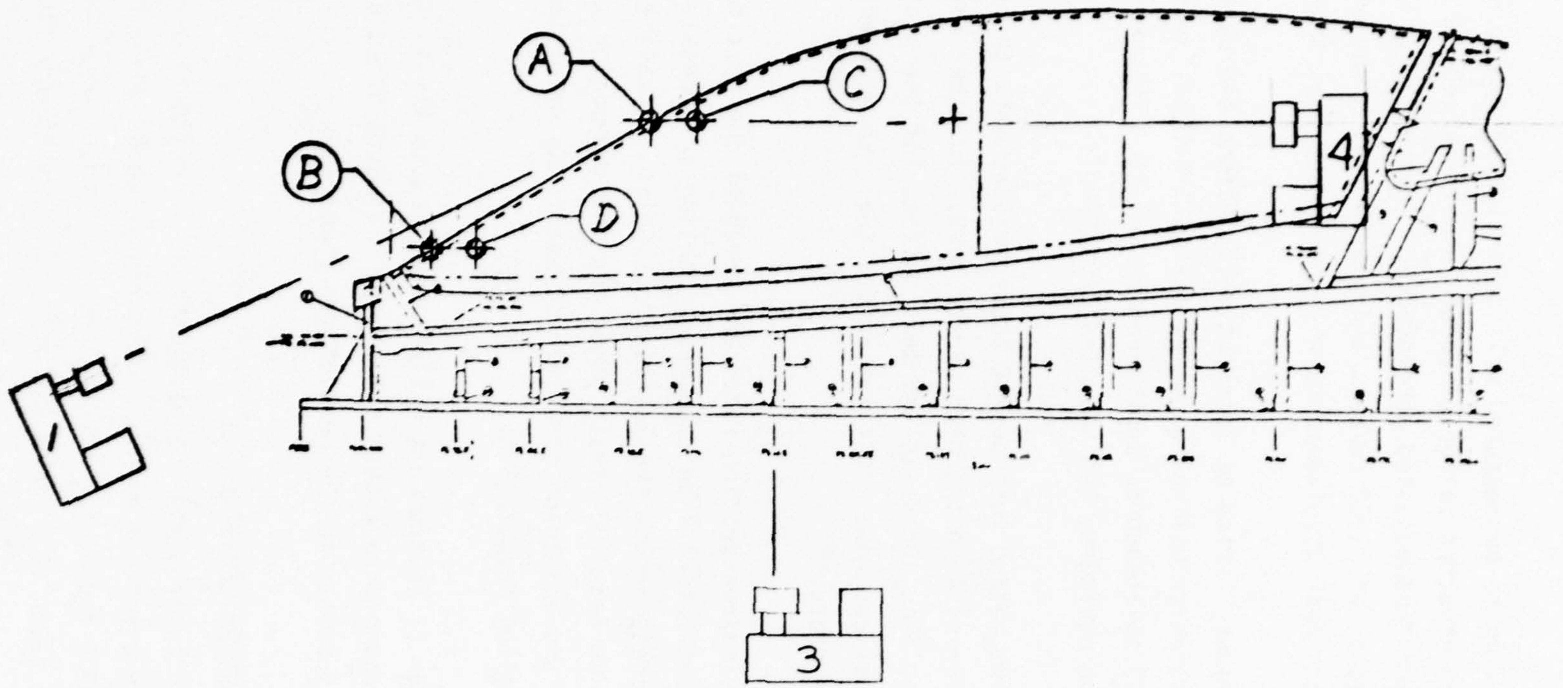


Figure A14. Camera Locations.

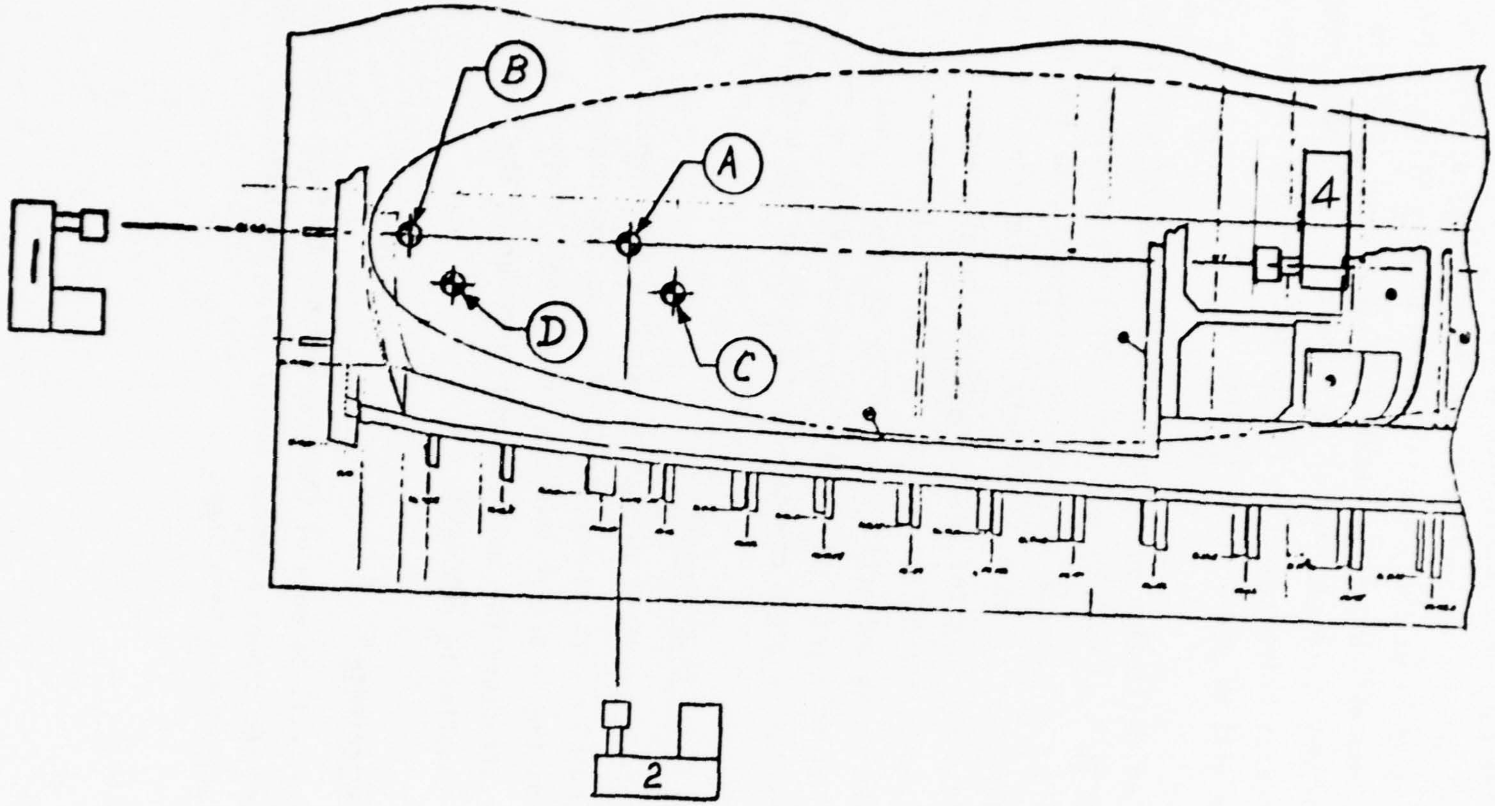


Figure A15. Camera Locations.

set up with appropriate angle of view and field to provide photographic information for the purpose of deflection measurement utilizing a grid system laid out on the inside surface of the transparency as shown in Figure A16. The grid lines shall consist of Dow Corning Silastic RTV-732 Silicone Sealant and will be approximately 0.25 inch wide. The grid pattern locations will be documented on Data Sheet 4. In order to reproduce the grid on each transparency in a quick and consistent manner, a Mylar template will be prepared at AEDC/ARO.

Nine points of deflection on the canopy centerline and nine points of deflection at BL 6.0 shall be measured, noting the appropriate scale factor. The points of measurements are the intersection of the grid lines and shall be labeled as shown in Figure A16. The time from plus gate to the reference origin shall be noted at all reference origin zero's. Deflections may be taken at one millisecond intervals. It is noted that during impact all target point deflections may not be attainable, but sufficient definition will be obtained from those which are in view. A fifth camera may be necessary for complete coverage of the side of the transparency.

Anthropomorphic Dummy Instrumentation

Networks, power supplies, balancing, sensitivity controls, signal conditioning and FM recording equipment to record six channels of acceleration recorded on the head/neck form and six channels of force measured on the support structure for the head/neck form during the qualification test shall be supplied by AEDC/ARO.

Time Study Requirements

A time study will be made of the transparency to frame installation and removal and the canopy assembly installation and removal. Pertinent data will be recorded on Data Sheet 1.

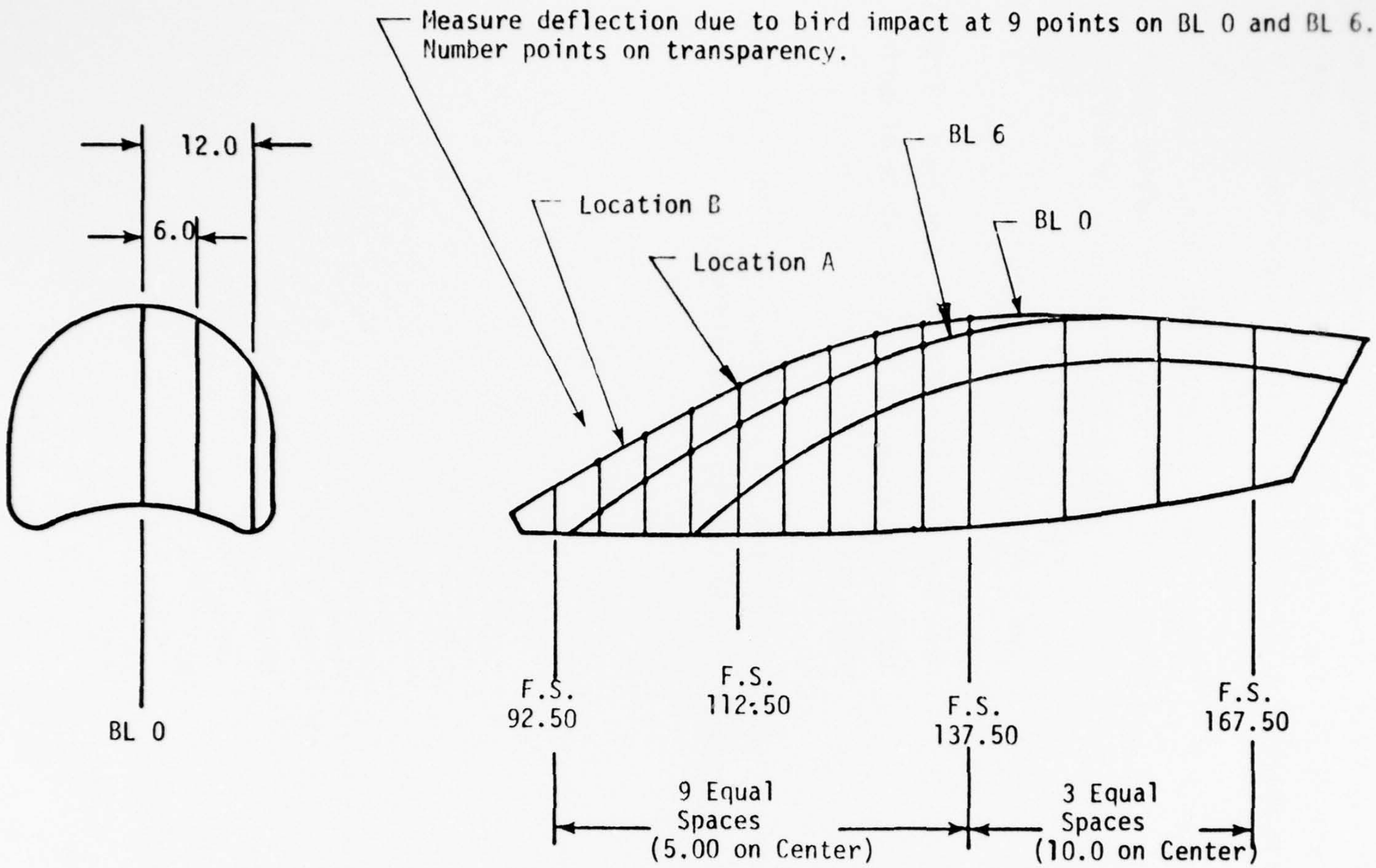


Figure A16. Grid Pattern on Inner Surface of Transparency. (Tolerances are $\pm$ 0.12).

Instrumental Installation Instructions

Strain gages and thermocouples shall be bonded to their respective surfaces with GA-2, or equivalent, adhesive per Micro-Measurements Instruction Bulletin B-137-3 with the following exceptions:

Paragraph III	Step 1 - Do not use chlorothene nu.
	Step 2 - Use 400 grit silicone carbide paper in place of 220 or 320. Do not use conditioner.
	Step 3 - Do not use neutralizer.
Paragraph IV	Cure two hours at 150°F to insure bond. Post cure during panel curing cycle at panel curing temperature.

Confine preparation and cement to the gage area. Use clean room procedures in handling and installing the strain gages.

TEST PROCEDURE

The tests shall be conducted in the following sequence per the test schedule and requirements given in Table A1. Revisions may be made at the discretion of the Douglas Test Director.

Phase I

1. Install the canopy frame and the table top test fixture to the target platform.

2. Install Transparency Number C1 to the canopy frame for Static Test Number 001. Record the time and sequence to install on Data Sheet 1.

3. Attach all instrumentation wires from the transparency and structure to the recording apparatus. Complete the instrumentation checklist, Tables A3 and A4, and perform the calibration tests on the recording apparatus.

4. Install the nine deflectometers and record the locations on Data Sheet 15 and 16. Locate a static load hydraulic system in an appropriate position to impact the transparency at Location A, Figure A5. Install a thermal curtain, Figure A10.

5. Adjust the transparency temperature per Table A1, and start the instrumentation.

6. Perform Static Test Number 001 at 75°F by applying a static load at Location A of 2500 pounds or 1.00 inch deflection, whichever comes first.

7. Record deflection data, strain gage data, thermocouple readings, sensing element readings, relative humidity, ambient temperature, chamber temperature, and cockpit temperature.

TABLE A3. INSTRUMENTATION CHECKLIST - PREPARATIONS

A.	Record the transparency grid pattern on Data Sheet 2. Verify the location of the impact points on the transparency and record the location on Data Sheets 3 and 4.
B.	Strain gages: Check continuity, measure resistance at terminals, identify channel by channel, identify proper scaling for tape records, and identify sensitivity settings and shunt calibrations. Complete Data Sheets 5, 6, 7 and 8. Prepare sheet, referencing each strain gage channel to recorder channel; include calibration resistor value and equivalent in strain.
C.	Set up log for recording each event on oscillograph or tape; mark on records where possible; record clock time (hours/minutes) of events on the shot log.
D.	Set up temperature recorder; check identity of each channel; include calibration resistor value and equivalent in strain. Complete Data Sheets 9 and 10.
E.	Equipment in Position: Obtain still photos before every test, showing camera positions and strain gage details. As accurately as possible determine position in space of camera lens angles with canopy, etc. Record camera locations on Data Sheets 11 and 12. Perform scale calibration of movie camera for all cameras. Provide reference position markers (crosses, dots, etc.) for movies and a plus gate marker on movies for deflection data correlation with time.
F.	Obtain relative humidity reading from local source. Prepare Data Sheets 13 and 14.
G.	Provide DAC with measurement system schematic showing identification, serial numbers, cable type and numbers, signal conditioner type and numbers, setting for equipment sensitivities and equipment calibration dates. (May be completed after test.)

TABLE A4. INSTRUMENTATION CHECKLIST - ENVIRONMENTAL PRE-TEST

A. Take zero record on tape/recorders.

B. Take calibration record on tape/recorders.

C. Start temperature recorder and continuously record or incrementally time record if applicable during heating or cooling.

D. Incrementally time record (intervals TBD) strain gage channels on tape recorders. Correlate with temperature records.

E. When target temperatures are reached and stabilized, take tape/recorders recordings.

8. Quickly release the load and record the strain gage data during specimen damping.

9. Retract the hydraulic cylinder to its original position. Examine the strain gages, thermocouples, and deflectometers for damage. Refurbish and/or replace any damaged units. Examine the transparency and support structure for possible damage. Evaluate damage and proceed if acceptable.

10. Perform Static Test No. 002 at 195°F on Transparency Number C1 by repeating Steps 3 through 9.

11. Perform Static Test No. 003 at -35°F on Transparency Number C1 by repeating Steps 3 through 9.

12. Remove the deflectometers and prepare for Test Number 004.

13. Attach all instrumentation lead wires from the transparency and structure to the recording apparatus, complete the instrumentation checklist, Tables A3 and A4, and perform the cali- proper calibration tests on the recording apparatus. Position the support structure and gun to impact the transparency at the proper location per Table A1. Install a thermal hood, Figure A4 over the canopy assembly and support structure when necessary to maintain the appropriate temperature, Table A1.

14. Complete the instrumentation checklist, Table A5, adjust the transparency temperature per Table A1, start the instrumentation, take static strain gage readings immediately prior to dropping the thermal curtain, and drop the thermal curtain prior to firing.

TABLE A5. INSTRUMENTATION CHECKLIST - BIRD SHOT

RECORD EVENT TIME OF EACH STEP.

A. Prepare gun for firing. Trim and weigh chicken.

B. Take zero record on tape/recorders.

C. Take calibration record on tape/recorders.

D. Monitor and record electrical sensing elements before test.

E. 1. Start count down.

a) Turn on temperature recorder T-30 seconds, <u>Mark</u> at T-0.

b) Turn on oscillograph at T-5 seconds, 80 inch/second (if applicable).

c) Turn on tapes T-5 seconds, 120 inch/second.

d) Mark T-lights on and T-0 and T-15 seconds.

e) Turn on cameras T-1 second.

f) Mark time correlation at T- 1 second.

2. <u>Fire</u> - Note time of firing.

15. Perform Test Number 004 by impacting Transparency C1 at Location A with a four pound bird traveling at a velocity of 350 knots.

16. Record the appropriate data on data sheets in accordance with the complete instrumentation checklist, Table A6.

17. Visually check the transparency, C1, the thermocouples, and the structure for damage. Evaluate the damage, repair as necessary, and proceed to Step 18 if the transparency is acceptable. If Transparency C1 is penetrated or damaged, replace C1 with C4 and proceed to Step 21.

18. Prepare for Test Number 005 by completing Steps 13 and 14 for C1.

19. Perform Test Number 005 by impacting Transparency C1 at Location C with a four pound bird traveling at a velocity of 350 knots. Repeat Step 16.

20. Visually check Transparency C1, the thermocouples, and the structure for possible damage. Evaluate the damage, repair, and proceed if C1 is usable. If the transparency is damaged, replace C1 with C4 and proceed.

Phase Ib

21. Prepare for Test Number 006 by completing Steps 13 and 14, except strain gage data is not required.

22. Perform Test Number 006 by impacting Transparency C1 (or C4) at Location A with a two pound bird traveling at a velocity of 350 knots. Complete Step 16.

TABLE A6. INSTRUMENTATION CHECKLIST - POST TEST

A. Tabulate data on Douglas Data Sheets 13 and 14, and obtain test signatures.
B. Obtain still photos of damage and condition of specimen.
C. Strain gage data may be reviewed quick-look by playback on oscillograph. Check for scaling and qualitative results. Also, compare with digitized data for timing and magnitude.
D. Data from digitizer and plotter to be available for review within 48 hours (turnaround time expected to improve for future tests). (Watch out for scaling errors.)
E. All strain gage data must be plotted versus time from plus gate to the same reference origin.
F. Review movies and make notes.
G. Assist in and obtain deflection data from optical reading system and computer program.
H. Provide suggestions/changes for improving next test.

23. Complete Step 14 and perform Test Numbers 007, 008, 009, and 010 by firing two pound birds at Location A at the velocities shown in Table A1, or until the transparency fails. A failure will terminate Phase Ib. Complete Step 16.

24. Replace Transparency Number C1 (or C4) with C2. Record the sequence and time to remove and install.

Phase Ic

25. Prepare for Test Number 011 by completing Steps 13 and 14.

26. Perform Test Number 011 at 195°F by impacting Transparency C2 at Location A with a four pound bird traveling at 350 knots. Complete Step 16.

27. Visually check the transparency, C2, the thermocouples, and the structure for damage. Evaluate, repair, and proceed if C2 is usable. If the transparency is damaged, replace C2 with C5.

Phase Id

28. Prepare for Test Number 012 by completing Steps 13 and 14, except strain gage data is not required.

29. Perform Test Number 012 by impacting Transparency C2 (or C5) at Location A with a three pound bird traveling at 350 knots. Complete Step 16.

30. Repeat Step 14 and complete Test Number 013, 014, 015, and 016 by firing at Location A three pound birds at the velocities shown in Table A1, or until the transparency

fails. A failure terminates Phase Id. Complete Step 16.

31. Replace Transparency C2 (or C5) with C9. Record the time to remove and install and the sequence of events.

Phase II

32. Perform Test Number 017, Phase II.

33. Replace Transparency C9 with C3. Record the time to remove and install.

Phase Ie

34. Prepare for Test Number 018 by completing Steps 13 and 14.

35. Perform Test Number 018 at -35°F by impacting Transparency C3, at Location A, with a four pound bird traveling at 350 knots. Complete Step 16.

36. Visually check Transparency C3, the thermocouples, and the structure for possible damage. Evaluate, repair, and proceed if C3 is usable. If C3 is damaged and not usable, replace C3 with C4 or C5. Phase Ib and Id have a higher priority than Test Numbers 019 and 020. Transparency Number C4 or C5 may be used for Test Number 019 only when C1 has been used to complete Phase Ib or C2 has been used to complete Phase Id.

37. Prepare for Test Number 019 by completing Steps 13 and 14 except strain gages are not required. The glare shield and instrument panel, which had been used for Phase II, Test Number 017, will be installed to the table top test fixture for Test Number 019.

38. Perform Test Number 019 at 75°F by impacting Transparency C3 at Location B with a four pound bird traveling at a velocity of 350 knots. Complete Step 16.

39. Visually check the transparency, the thermocouples, and the structure, especially the canopy forward frame, for damage. If the frame is damaged, repair or replace with the canopy frame from the Phase II test. Repair the glare shield if necessary. If the transparency is penetrated or damaged, Test Number 020 will be deleted. Proceed to Step 42. If the transparency is usable, proceed to Test Number 020.

40. Prepare for Test Number 020 by completing Step Numbers 13 and 14. The glare shield and instrument panel will be installed.

41. Perform Test Number 020 at 75°F by impacting Transparency C3 (C4 or C5) at Location D with a four pound bird traveling at a velocity of 350 knots. Complete Step 16. Remove the transparency and record the time to remove.

Phase If

42. Prepare for Static Test Number 021 by installing Transparency Number 021. Record the time and sequence to install.

43. Complete Static Test Number 021 at 75°F on transparency by completing Steps 3 through 9.

44. Remove the deflectometers and prepare for Test Number 022 by completing Steps 13 and 14. The glare shield, instrument panel, and a simulated HUD, consisting of side frames and glass, will be installed.

45. Perform Test Number 022 at 195°F by impacting C6 at Location A with a four pound bird traveling at 350 knots. Complete Step 16.

46. Check the transparency, the thermocouples, and the structure for damage. Evaluate, repair, and proceed to Test Number 023, Step 47, if Transparency C6 is usable. If C6 is damaged, remove and replace with Transparency C7 and proceed to Step 51.

Phase Ig

47. Prepare for Test Number 023 by completing Steps 13 and 14 for Transparency C6. Strain gage data is not required. The instrumented panel, glare shield, and simulated HUD will be installed.

48. Perform Test Number 023 at 75°F by impacting Transparency C6 at Location A with a four pound bird traveling at 350 knots. Complete Step 16.

49. Repeat Step 14 and complete Test Numbers 024, 025, 026, and 027 by firing four pound birds at Location A (the location may be revised at the discretion of the Douglas Test Director) at the velocities shown in Table 1, or until the transparency is penetrated or is deemed to be unusable. A failure terminates Phase Ig. Complete Step 16.

50. Replace Transparency C6 with C7. Record time to remove and install.

Phase Ih

51. Prepare for Test Number 028 by completing Steps 13 and 14. The glare shield, instrumented panel, and simulated HUD assembly will be installed.

52. Perform Test Number 028 at -35°F by impacting Transparency C7 at Location B with a four pound bird traveling at a velocity of 350 knots. Complete Step 16.

53. Check the transparency, the thermocouples, and structure for damage. If Phase Iq, Steps 47, 48, and 49, have been completed, the basic test schedule is finished. If Phase Iq has not been completed (See Step 46), proceed to Step 54.

54. If the frame is damaged, repair or replace with the canopy frame from Phase II, Test Number 017. If Transparency C7 has been penetrated or damaged, replace Transparency C7 with C8.

55. Prepare for Test Number 023 by completing Steps 13 and 14 for Transparency C7 or C8.

56. Perform Test Numbers 023 through 027 by completing Steps 47, 48, and 49. Complete Step 16.

Phase II

57. Test Numbers 029 through 034 will be completed at the discretion of the Douglas test director by following the procedures established in the preceeding steps of this section. These tests may be accomplished if Transparency C4, C5, C7, or C8 are usable.

DOCUMENTATION

Douglas will provide forms for the static loading/dynamic unloading tests and the bird impact test to record those items that require direct readings of instrumentation before, at-time-of, and after the tests (including channel identification). Such data will include temperature and sensing element readings and other physical test parameters. See Data Sheets 13 and 14.

AEDC is to supply oscillograph readings clearly identified regarding scale and calibration. The equipment used to record data from the strain gage channels shall be sufficient to supply strain gage data presented in Engineering units, μ inch/inch versus time from plus gate. The period of interest will include the duration of impact, which will be about half a millisecond, and some 5 to 10 milliseconds subsequent to impact.

AEDC shall furnish one inch width FM tape recordings of the strain gage channels and the IBM cards used to generate the deflection data, including scale factors.

Figure A17 shows a suggested format for a rubber stamp or decal which should be applied to oscillograph rolls, tape or other records/data as soon as that record is taken from its generating machine. The information should then be filled out by the machine operator promptly. This information is the minimum required to properly identify any data and provide maximum traceability and permanent documentation.

Data reduction requirements include:

- Strain gage data digitized and plotted versus time from plus gage. All data to be plotted must start at the same reference origin time. The format for data presentation will be as mutually agreed and the acquisition will be the

responsibility of AEDC. The format absolutely required will be the strain gage data presented in engineering units, 1000 inch/inch per division, versus time correlated with plus gate to initial impact (printed data and plots). A sample strain/time plot is shown in Figure A18. AEDC will provide these data to Douglas.

- Deflections digitized and plotted as
  a) Vertical displacement versus horizontal displacement
  b) Maximum displacement ($x \rightarrow y$) versus time from plus gate.

- Temperature recording trace and/or tapes.

BIRD IMPACT TESTING

DATE ____________

DATA NO. TEST ____________________ TIME ____________

TEST PROCEDURE NO. ____________________

INSTRUMENTATION TYPE ____________________

CHANNELS ____________________

INFO ____________________

____

MACHINE ________________ OPERATOR ____________

Figure A17. Recommended Format for Rubber Stamp or Decal for Oscillograph Strain Gage or Temperature Records.

This information should be made available for user review as soon as possible after the particular test.

AEDC to provide high-speed camera (5000 fr/sec) coverage to measure deflection versus time during the impact. Figures A14 and A15 show approximate camera locations. AEDC is to supply Douglas with each camera lens coordinates, camera viewing angle, and edited, unspliced copies of the films.

NOTES:

(1) T_0 refers to the time from plus gate to this reference origin.

(2) t refers to even intervals of time to be added to the reference origin in milliseconds.

(3) Plus gate refers to the pulse triggered on a recorded electrical trace when a current-carrying wire in the path of the bird breaks.

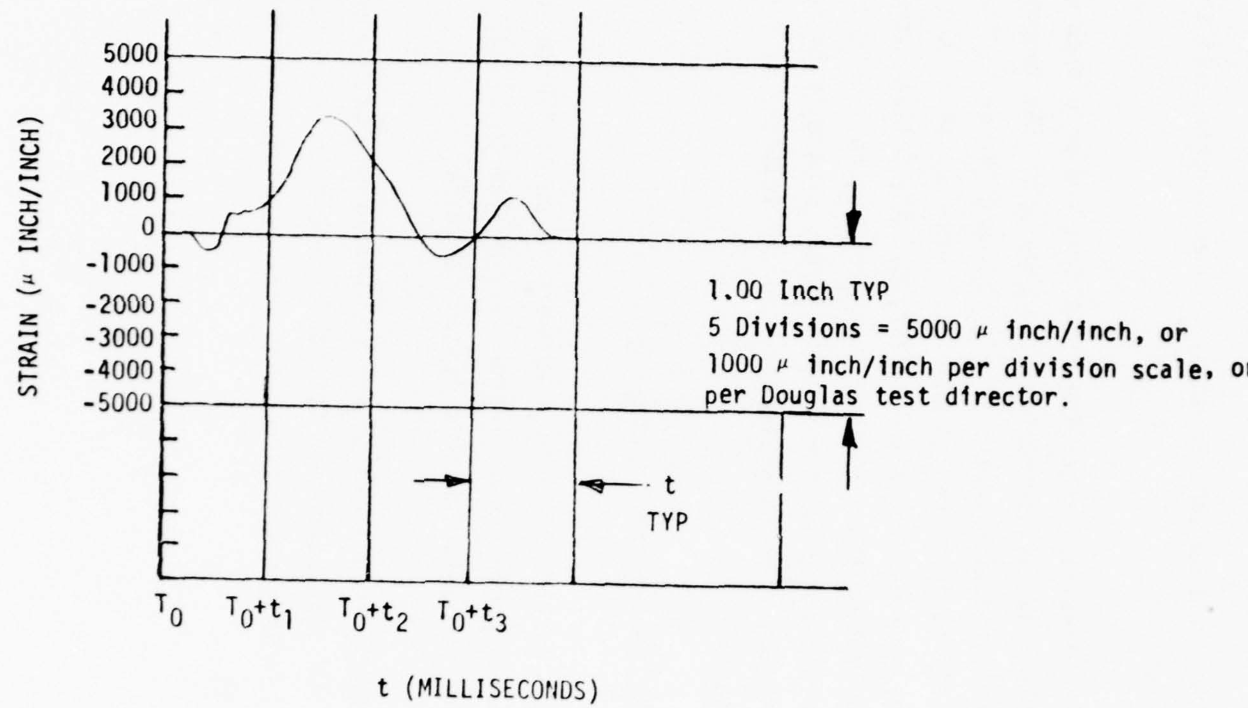


Figure A18. Sample Digitized Strain Plot Versus Time from Plus Gate.

Data shall be sent to Douglas in accordance with the following schedule:

> Five days after the first test and once each week thereafter the oscillograph strain gage data, the temperature data, the deflection versus time plots, and the deflection IBM cards shall be delivered to Douglas. Ten days after the first test and once each week thereafter, the digitalized strain data and deflection films shall be sent to Douglas. Good definition must be given of when the bird is fired, plus gate to impact time, the actual point of impact, and the velocity of the bird at impact.

DISPOSITION OF TESTED HARDWARE

Disposition of tested hardware is an ASD/YP responsibility. The AFFDL/FEW program coordinator will be the focal point for receipt of written requests for this tested hardware.

Douglas requirements are for Canopy C1, C2, C3 and C6 which are the four transparencies tested for the math model correlation. Douglas will remove coupons from these transparencies for material properties testing.

F-16 CANOPY TESTS

DACO DATA SHEET PACKAGE

TEST NO. ________________ CANOPY NO. ______________ S/N ________

Data Sheet No.	Information Type	Remarks
1	Time Study Data Sheet	
2	Grid Pattern on Inner Surface	
3	Bird Impact Target Point Locations	
4	Bird Impact Target Point Coordinates	
5	Strain Gage Usage Chart	
6	Strain Gage Locations - Canopy	
7	Strain Gage Locations - Fwd Canopy Frame	
8	Strain Gage Coordinates	
9	Thermocouple Locations	
10	Thermocouple Coordinates	
11	Camera Locations (Top View)	
12	Camera Locations (Side View)	
13	Canopy Bird Impact/Static Loading Test	
14	Thermocouple Readings	
15	Static Deflectometer Locations	
16	Static Deflectometer Coordinates	

DACO Representative ___________________________

DATA SHEET 1
TIME STUDY DATA SHEET

Canopy No. __________

Serial No.__________

Transparency Installation

Start Time: __________

End Time: __________

Comments: ____

____

Canopy Installation

Start Time: __________

End Time: __________

Comments: ____

____

Canopy Removal

Start Time: __________

End Time: __________

Comments: ____

____

Transparency Removal

Start Time: __________

End Time: __________

Comments: ____

____

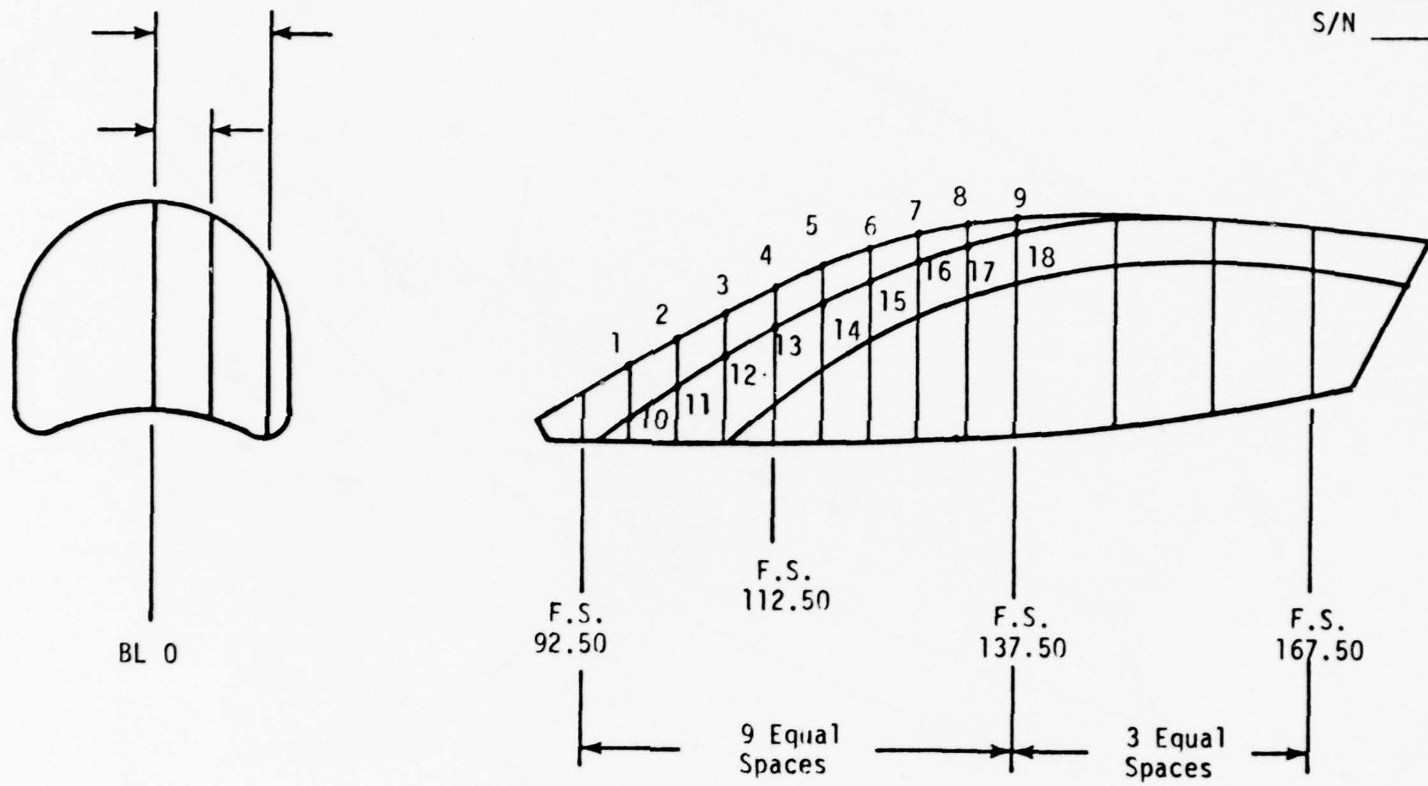
DATA SHEET 2

GRID PATTERN ON INNER SURFACE

Test No. _______

Canopy No. _______

S/N _______

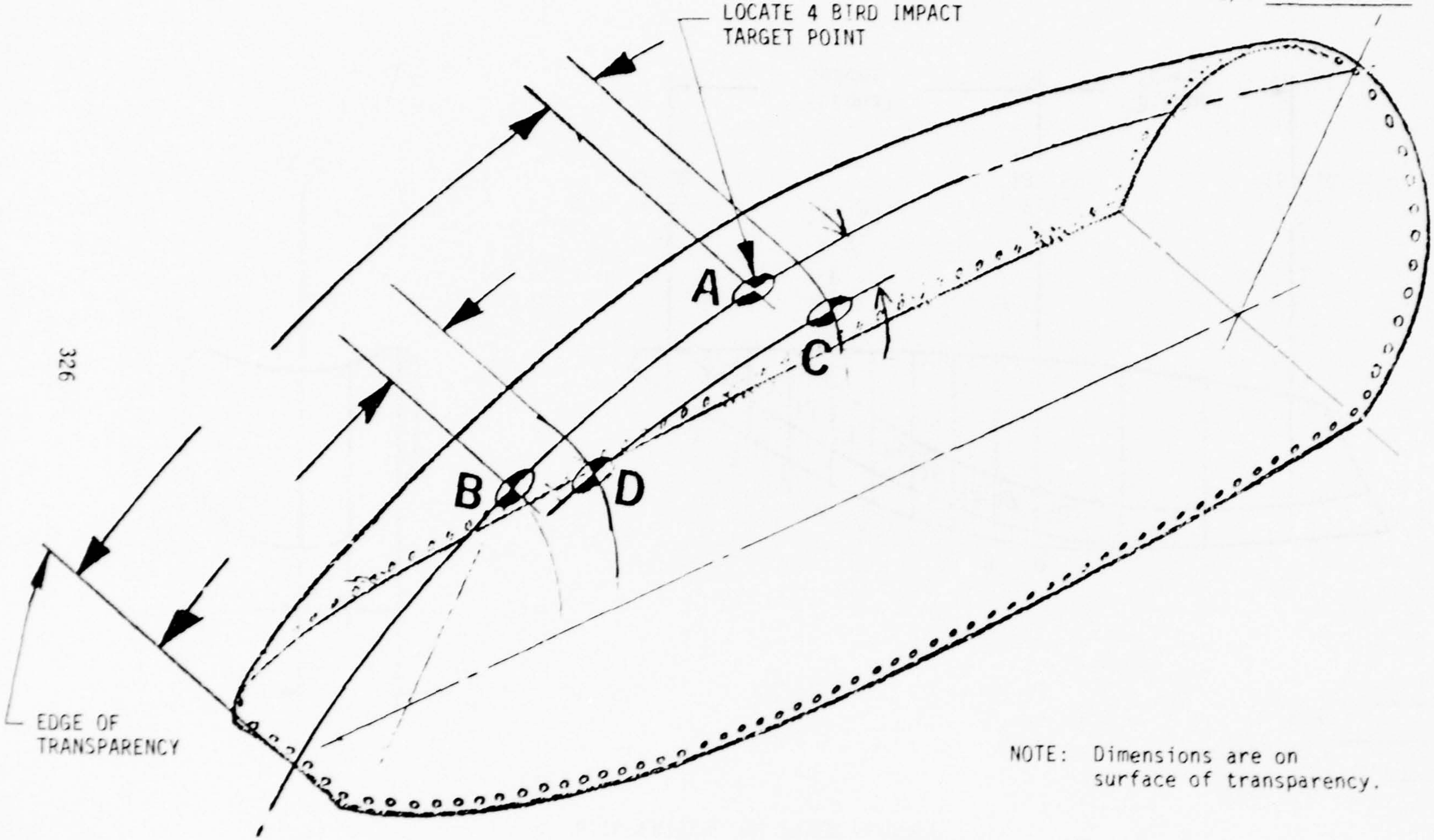


DATA SHEET 3
BIRD IMPACT TARGET POINT LOCATIONS

Test No. ____

Canopy No. ____

S/N ____



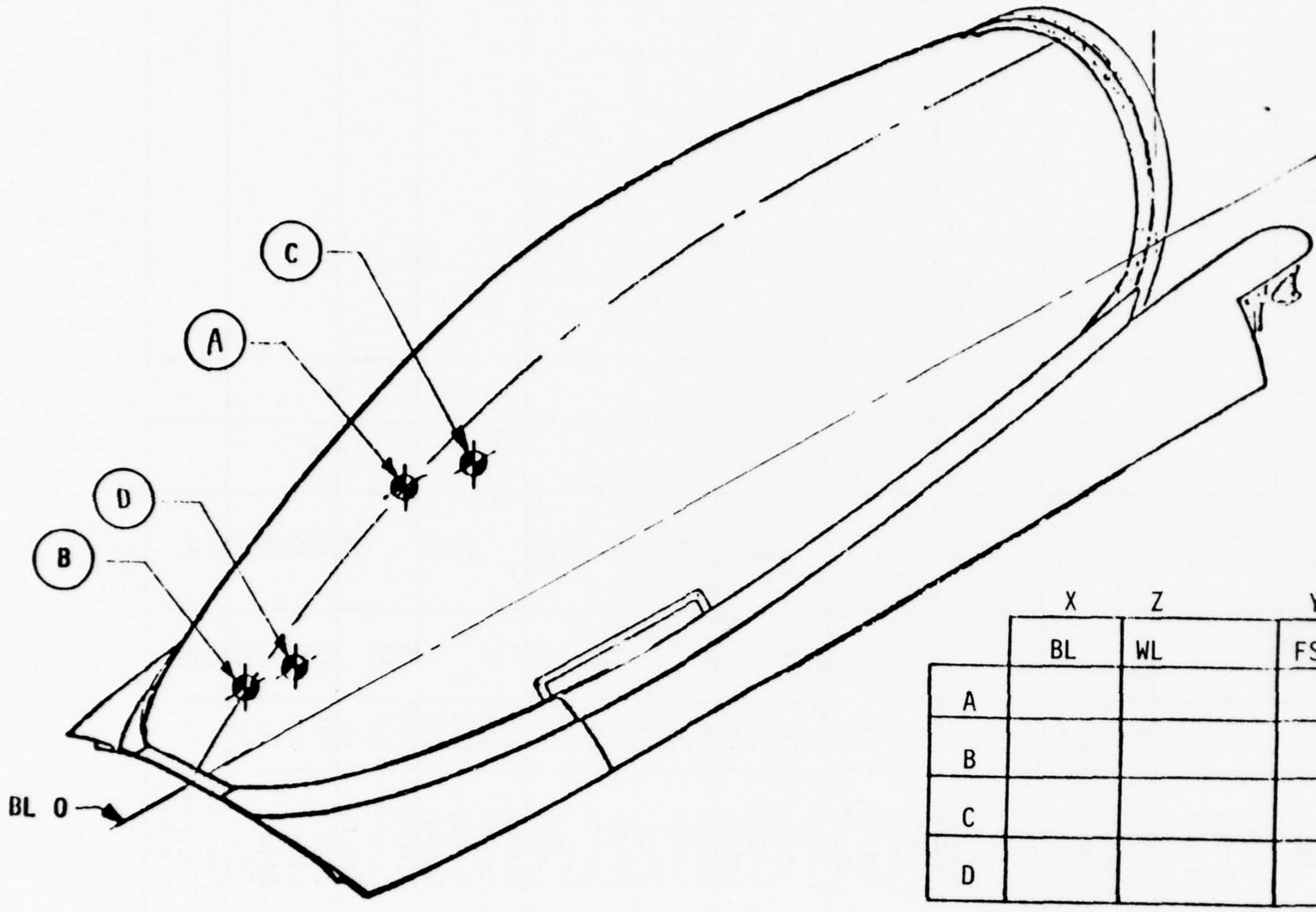
NOTE: Dimensions are on surface of transparency.

DATA SHEET 4
BIRD IMPACT POINT COORDINATES

Test No. ________

Canopy No. ________

S/N ________



	X	Z	Y
	BL	WL	FS
A			
B			
C			
D			

NOTE: Locate origin on this data sheet.

DATA SHEET 5
STRAIN GAGE USEAGE CHART

Test No. _______

Canopy No. _____

S/N ___________

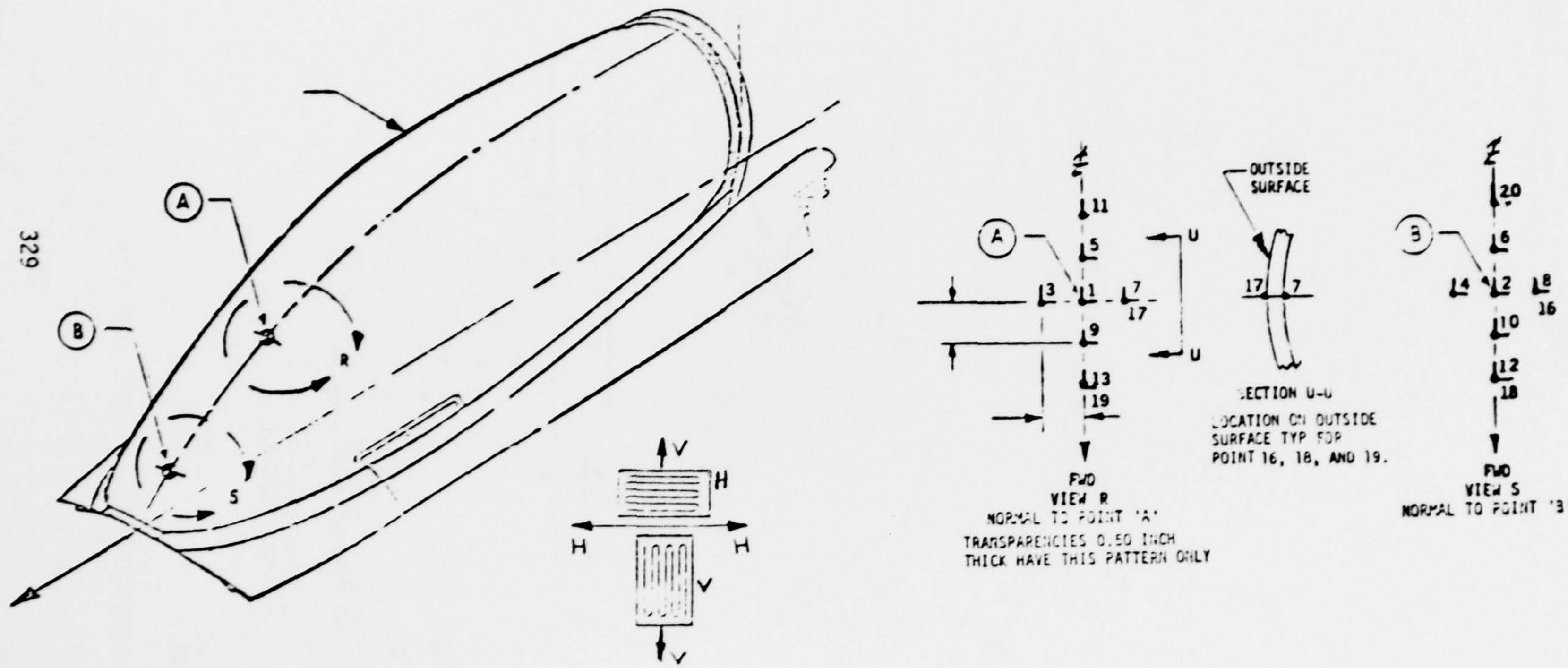
STRAIN GAGE LOCATION			CHANNEL SEQUENCE NO.				
LOCATION NO. & GAGE DIR.	MOUNTED ON	SIDE	SHOT LOCATION A	B	GAGE YES	OP. NO	REMARKS
1 H	Canopy	In	1				
1 V	Canopy	In	2				
2 H	Canopy	In		1			
2 V	Canopy	In		2			
3 H	Canopy	In	3				
3 V	Canopy	In	4				
4 H	Canopy	In		3			
4 V	Canopy	In		4			
5 H	Canopy	In	5				
5 V	Canopy	In	6				
6 H	Canopy	In		5			
6 V	Canopy	In		6			
7 H	Canopy	In	7				
7 V	Canopy	In	8				
8 H	Canopy	In		7			
8 V	Canopy	In		8			
9 H	Canopy	In	9				
9 V	Canopy	In	10				
10 H	Canopy	In		9			
10 V	Canopy	In		10			
11 V	Canopy	In	11				
12 H	Canopy	In		11			
12 V	Canopy	In		12			
13 H	Canopy	In	12				
13 V	Canopy	In	13				
16 H	Canopy	Out		13			
16 V	Canopy	Out		14			
17 H	Canopy	Out	14				
17 V	Canopy	Out	15				
18 H	Canopy	Out		15			
18 V	Canopy	Out		16			
19 H	Canopy	Out	16				
19 V	Canopy	Out	17				
20 V	Canopy	In		17			
28 H	Edge	Out	18	18			
28 V	Edge	Out	19	19			
29 H	Edge	In	20	20			
29 V	Edge	In	21	21			
30	Fwd Hook		22	22			
31	Fwd Frame		24	24			
32	Fwd Frame						

DATA SHEET 6
STRAIN GAGE LOCATIONS - CANOPY

Test No. ________

Canopy No. ______

S/N ___________

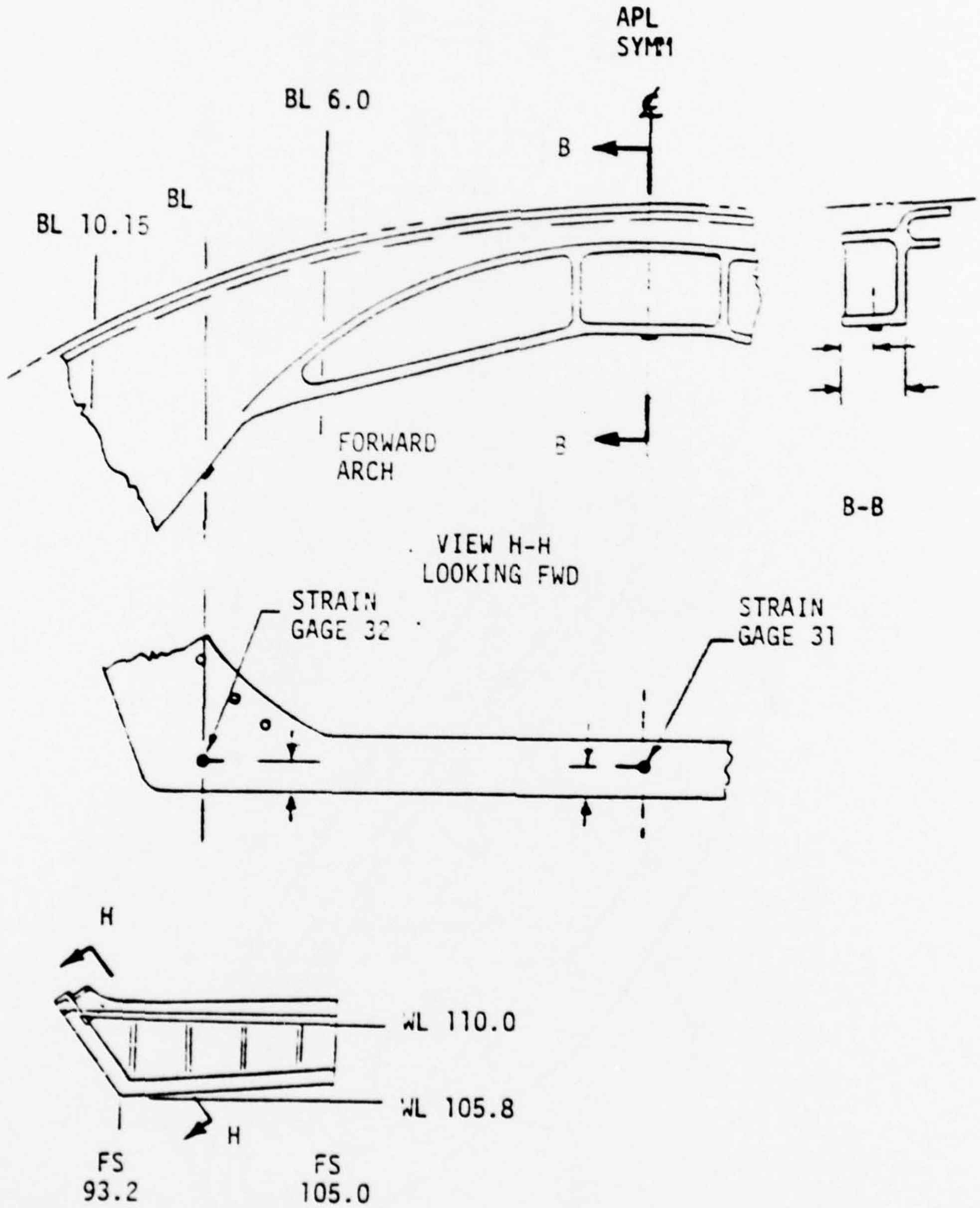


DATA SHEET 7
STRAIN GAGE LOCATIONS FORWARD CANOPY FRAME

Test No. ______

Canopy No. ____

S/N __________



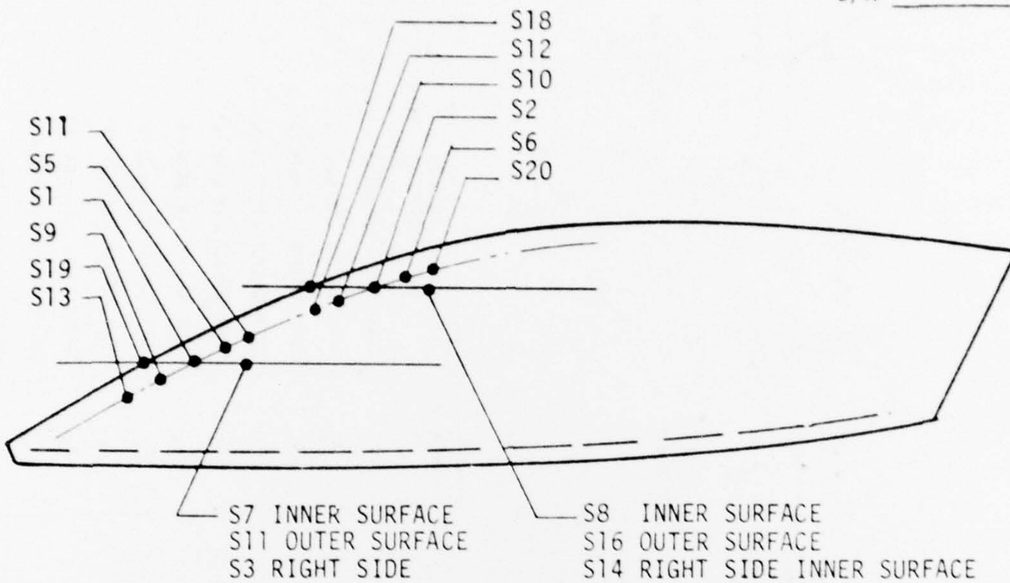
VIEW LOOKING INBOARD - LEFT HAND SIDE

DATA SHEET 8
STRAIN GAGE COORDINATES

Test No._______

Canopy No. ____

S/N ___________



	FS	WL	BL
S1			
S2			
S3			
S4			
S5			
S6			
S7			
S8			
S9			
S10			
S11			
S12			
S13			
S14			
S15			
S16			
S17			
S18			
S19			
S20			

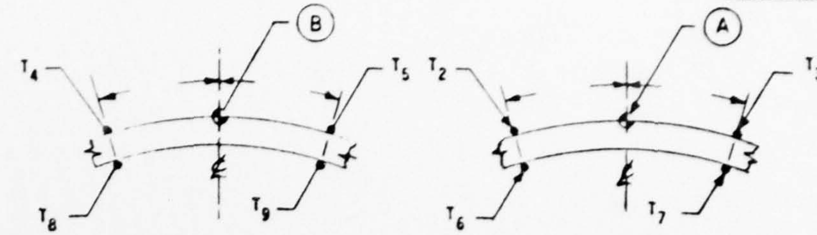
DATA SHEET 9

THERMOCOUPLE LOCATIONS

Test No. ________

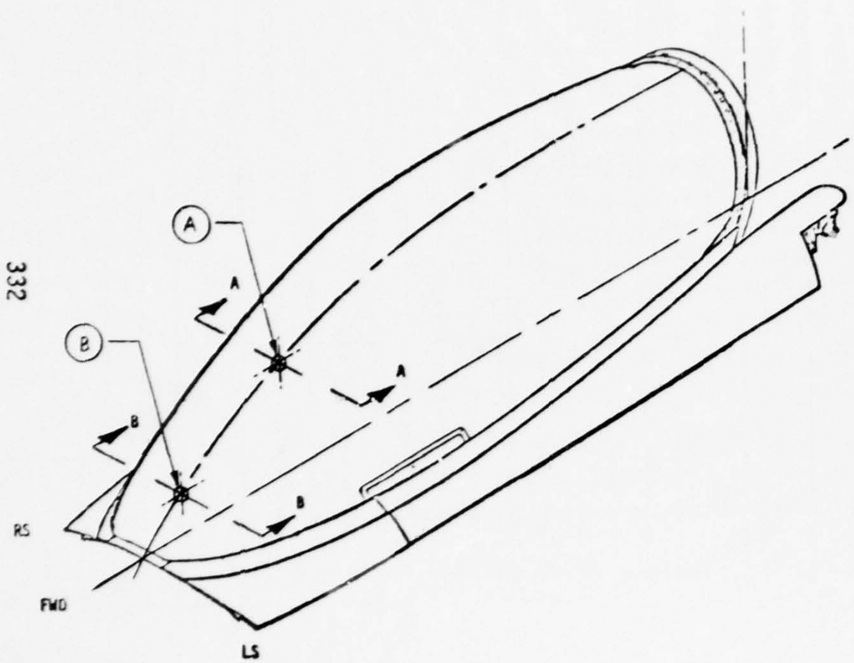
Canopy No. ________

S/N ________



SECTION B-B

SECTION A-A



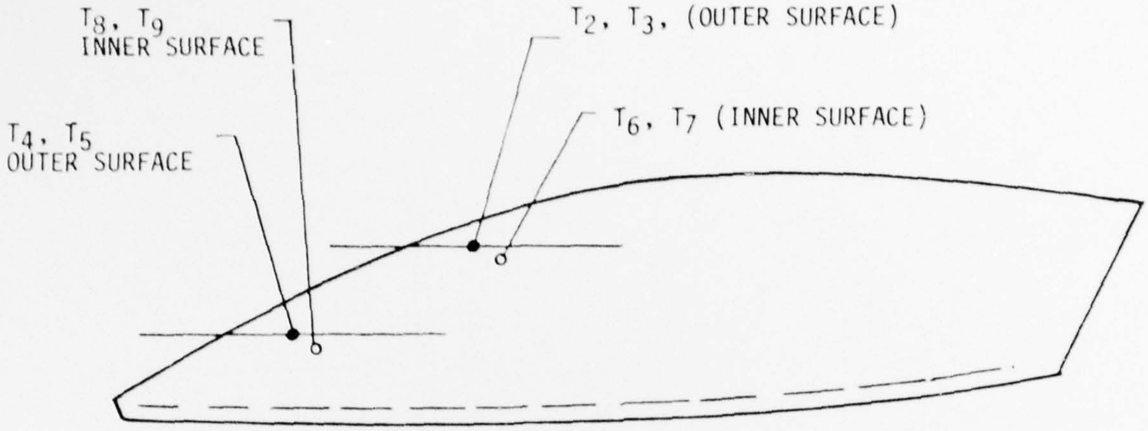
THERMOCOUPLE NO.	LOCATIONS FOR LOCATIONS OF A AND B
T_1	CHAMBER AIR TEMPERATURE - 3" ABOVE 'A'
T_2	OUTER SURFACE - RS NEAR 'A'
T_3	OUTER SURFACE - LS NEAR 'A'
T_4	OUTER SURFACE - RS NEAR 'B'
T_5	OUTER SURFACE - LS NEAR 'B'
T_6	INNER SURFACE - RS NEAR 'A'
T_7	INNER SURFACE - LS NEAR 'A'
T_8	INNER SURFACE - RS NEAR 'B'
T_9	INNER SURFACE - LS NEAR 'B'
T_{10}	COCKPIT AIR TEMPERATURE - 3" BELOW 'A'
T_{11}	AMBIENT AIR TEMPERATURE

DATA SHEET 10
THEROMOCOUPLE COORDINATES

Test No. ______

Canopy No. ____

S/N ____________



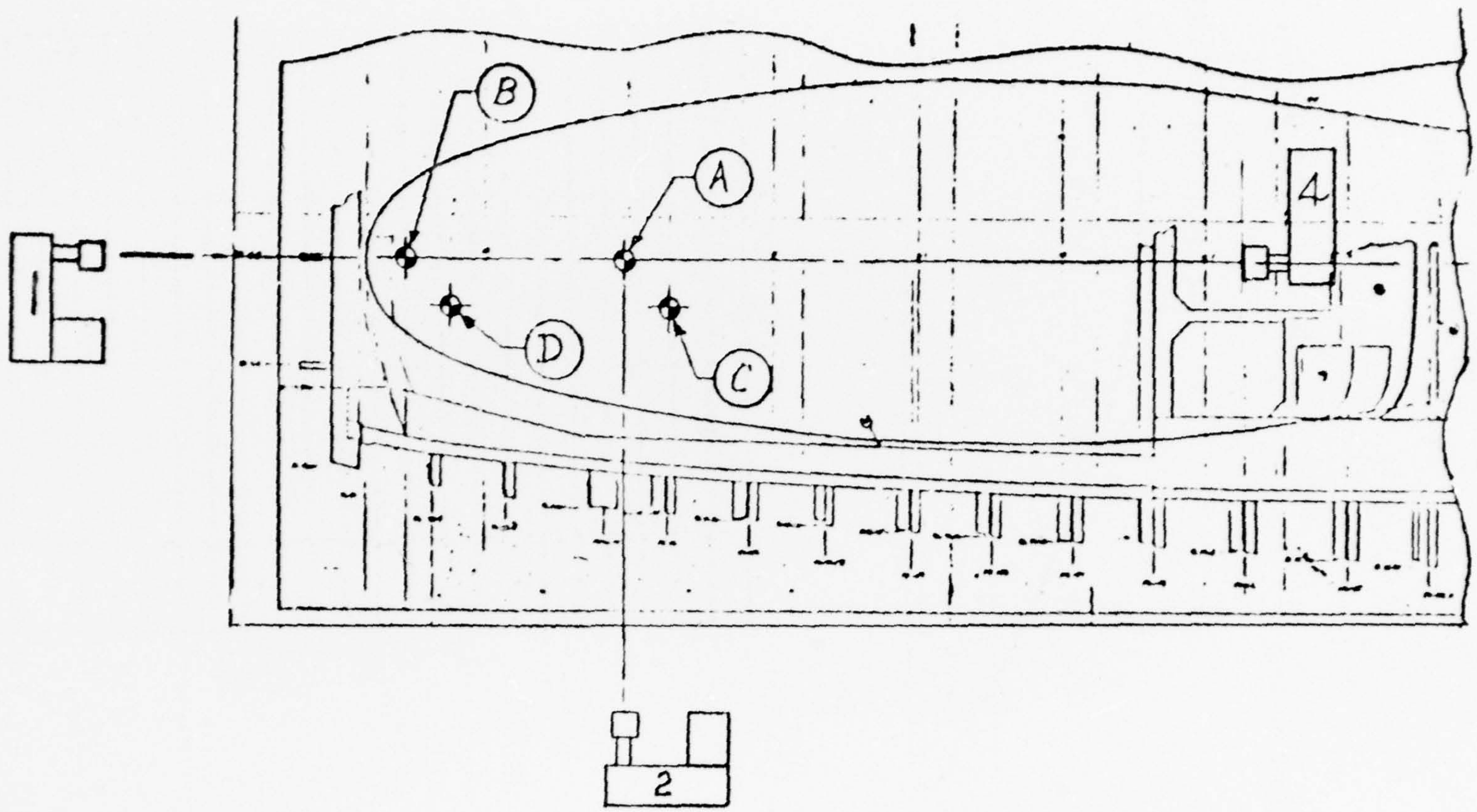
	FS	BL	WL
T_2			
T_3			
T_4			
T_5			
T_6			
T_7			
T_8			
T_9			

DATA SHEET 11
CAMERA LOCATIONS (TOP VIEW)

Test No. ________

Canopy No. ______

S/N ____________

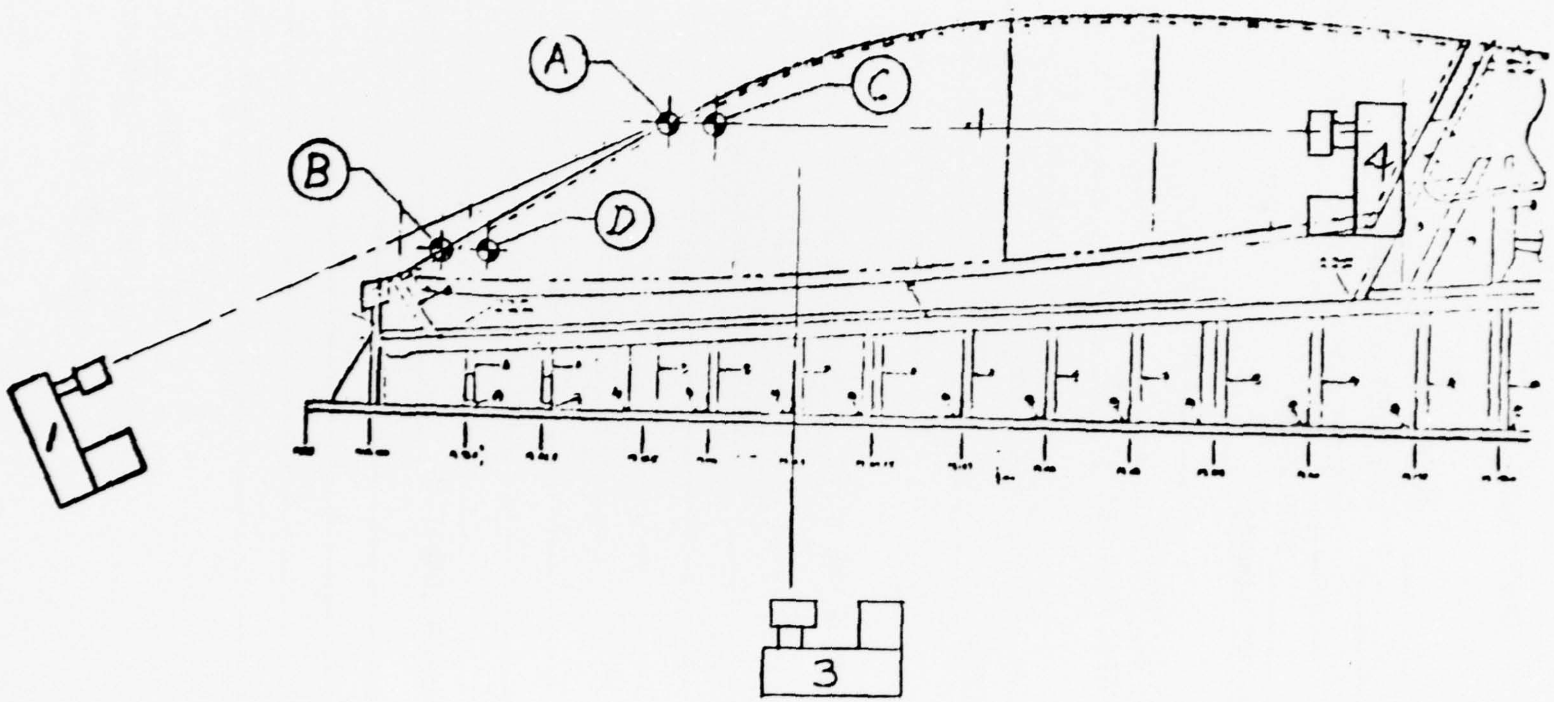


DATA SHEET 12
CAMERA LOCATIONS (SIDE VIEW)

Test No. ________

Canopy No. ____

S/N ____________



DATA SHEET 13

CANOPY BIRD IMPACT/STATIC LOADING TEST

TEST NO. ________________ CANOPY NO. ________________ S/N ________________

TEST CONDITION __________ LOAD/SHOT LOCATION __________ DATE ________________

RELATIVE HUMIDITY __________ START TIME __________ SHOT TIME ________________

BIRD WEIGHT ____________ BIRD VELOCITY ____________

RECORDER TAPE RECORD NUMBERS ____________

____

OSCILLOGRAPH RECORD NUMBERS ____

MOVIE FILM IDENTIFICATION C1 __________ C2 __________ C3 __________ C4 __________

FILM SPEEDS C1 __________ C2 __________ C3 __________ C4 __________

MAXIMUM DEFLECTION AT TARGET LOCATION ____

STATIC LOAD APPLIED ______________# ERROR $\pm$ ____________________#

COMMENTS

SPECIMEN CONDITION BEFORE TEST: ____

____

____

____

____

DESCRIPTION OF IMPACT: ____

____

____

____

POST IMPACT SPECIMEN CONDITION: ____

____

____

____

____

TEST CONDUCTOR: ____________________

AIR FORCE REPRESENTATIVE: ____________________

DACO REPRESENTATIVE: ____________________

DATA SHEET 14

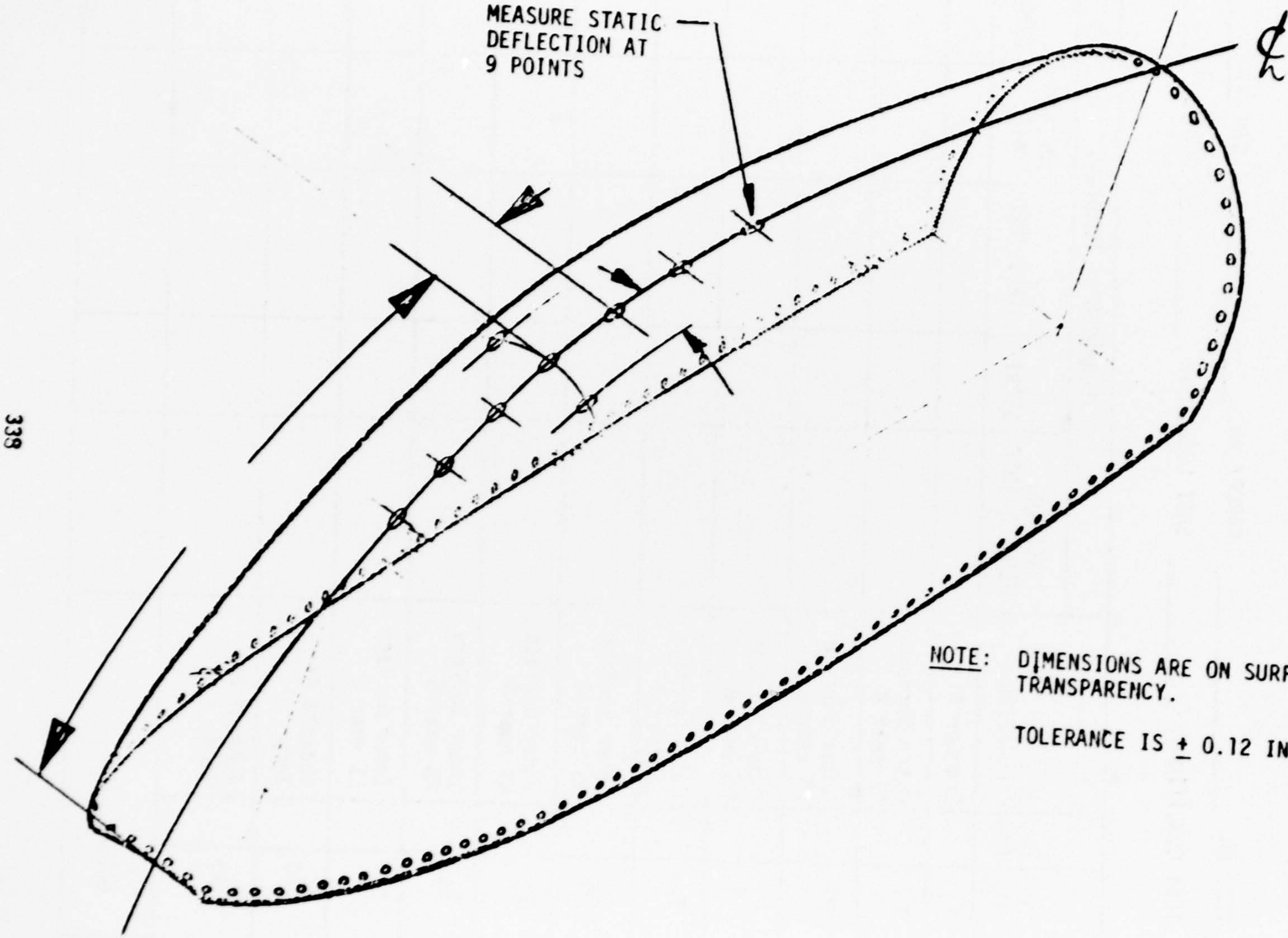
THERMOCOUPLE READINGS

TEST NO. ____________ CANOPY NO. ____________ S/N ____________

TEST CONDITIONS: ____________ SHOT LOCATION: ____________ DATE: ____________

TC NO.	CH. NO.	LOCATION	THERMOCOUPLE READINGS				
			REQUIRE TEMP. $\pm$ 10°F	START	STABILIZED	CURTAIN DROP	IMPACT
T_1		Chamber Air					
T_2		Outer Surface RS near A					
T3		Outer Surface LS near A					
T_4		Outer Surface RS near B					
T_5		Outer Surface LS near B					
T_6		Inner Surface RS near A					
T_7		Inner Surface LS near A					
T_8		Inner Surface RS near B					
T_9		Inner Surface LS near B					
T_{10}		Cockpit Air Temp					
T_{11}		Ambient Air Temp					
ACTUAL TIME							

DATA SHEET 15
STATIC DEFLECTOMETER LOCATIONS



NOTE: DIMENSIONS ARE ON SURFACE OF TRANSPARENCY.

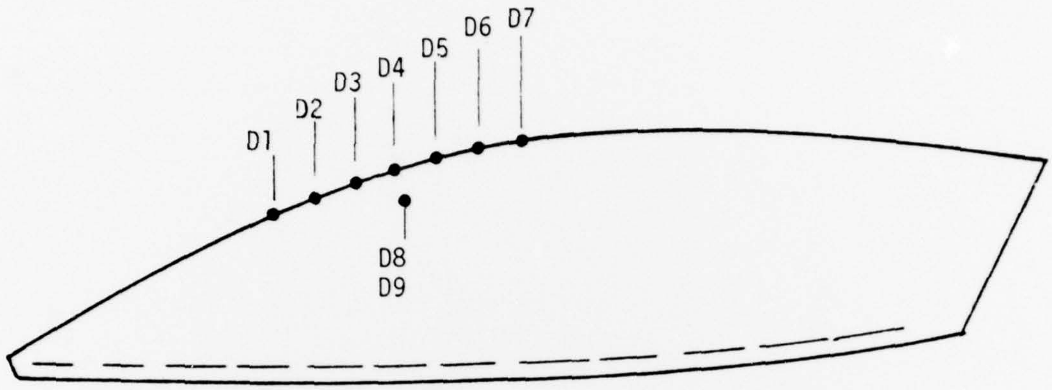
TOLERANCE IS ± 0.12 INCH

DATA SHEET 16
STATIC DEFLECTOMETER COORDINATES

Test No._________

Canopy No. ______

S/N ______________



	FS	WL	BL
D1			
D2			
D3			
D4			
D5			
D6			
D7			
D8			
D9			

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